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RESEARCH MEMORANDUM

INVESTIGATION OF THREE TAPERED 45° SWEEPBACK CAMBERED
AND TWISTED WINGS COVERING A SIMULTANEOUS VARIATION
IN ASPECT RATIO AND THICKNESS RATIO AND OF ONE
RELATED SYMMETRICAL WING AT TRANSONIC SPEEDS
BY THE WING-FLOW METHOD

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

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SUMMARY

An investigation has been made by the wing-flow method of three 45° sweptback cambered and twisted wings of taper ratio 0.25 which varied in aspect ratio and thickness ratio in such a manner as to represent closely practical-construction wings of approximately equal weight, equal strength, and equivalent aeroelastic properties. The wings had aspect ratios of 4, 6, and 8 and respective constant thickness ratios of 4, 8, and 12 percent of the chord and were all designed for a lift coefficient of 0.3 at a Mach number of 0.9. A fourth wing having the intermediate aspect ratio and thickness ratio but possessing no camber or twist was also tested to determine the effects of camber and twist at transonic speeds. The Mach number range was from 0.65 to 1.17 and the Reynolds number range was from approximately 230,000 to 620,000. In view of the low Reynolds numbers the results of this investigation must be considered preliminary until similar investigations are made at much higher Reynolds numbers.

Results showed that the lowest-aspect-ratio, thinnest wing tested had greater maximum lift coefficients, less drag, greater lift-drag ratios, and better stability characteristics than either of the other two higher-aspect-ratio, thicker, cambered and twisted wings. The effect of adding camber and twist to the intermediate aspect ratio wing was to decrease the maximum lift-drag ratios over most of the speed range tested. The data indicated that even though camber and twist reduced the drag due to lift, the accompanying penalty in zero-lift drag outweighed the beneficial effect. Camber and twist did, however, increase the maximum lift coefficient over the entire speed range tested and improved the variations of pitching moment with angle of attack or lift coefficient.

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INTRODUCTION

The designer of a modern high-speed swept-wing airplane is always faced with the question of what aspect ratio should be selected for the wing. In the age of low-speed, straight-wing airplanes, the answer to this question was not particularly difficult to find. If range and load-carrying ability were the most important considerations the designer chose a high-aspect-ratio wing which had low induced drag and consequently a high lift-drag ratio in the cruise condition. In order to reduce wing structural weight, the designer used large wing thickness ratios and was able to do this without paying a large penalty in increased profile drag at the speeds then common. If high speed and maneuverability were the goals, the designer chose a wing of somewhat lower aspect ratio inasmuch as wing strength and a small wing span were relatively more important and induced drag was relatively less important. Wing thickness ratios also generally were reduced but these reductions were tempered by the desires to keep the wing weight down and to maintain a high maximum lift coefficient.

With the advent of transonic and supersonic swept-wing airplanes, however, the problem of aspect-ratio selection becomes vastly different. Now, instead of being able to use a thick wing section with impunity, the designer finds that wing thickness ratio is perhaps the most powerful single determinant of total wing drag. Furthermore, the transition from straight to sweptback wings has brought with it many other problems that have a bearing on aspect-ratio selection — for example, the tendency of even rigid high-aspect-ratio swept wings to become unstable longitudinally at angles of attack far below the stall, and, as another example, the aeroelastic problem of loss in longitudinal stability of sweptback wings due to wing bending which was not important in straight-wing airplanes.

In order to stimulate further thought on this vital problem of wing aspect-ratio selection and to provide some preliminary results from small-scale experiments, the present investigation was made. An approximate theoretical analysis was made first to determine, for an assumed series of practical-construction wings, a logical variation of wing thickness ratio with wing aspect ratio. The results of this analysis were then applied to the construction of three solid steel wing models of constant sweepback (45°), constant taper ratio (0.25), with aspect ratios of 4, 6, and 8 and thickness ratios of 4, 8, and 12 percent of the chord, respectively. These models incorporated the calculated camber and twist required to make them support very nearly an elliptical span load distribution at $C_L = 0.3$ at $M = 0.9$. In order to investigate the effectiveness of the twist and camber, a fourth model of the intermediate aspect ratio and thickness ratio, but with a flat chord plane, was also designed and built. The remainder of the investigation

consisted of measuring the lift, drag, and pitching-moment characteristics of the series of four wings by the NACA wing-flow method. These tests covered a Mach number range from 0.65 to 1.17 and a Reynolds number range from about 230,000 to 620,000. Because of the low Reynolds numbers there is little likelihood that the data are applicable at full scale except perhaps for future aircraft operating at extreme altitudes; however, it is thought that some of the primary data trends may apply to wings of large scale inasmuch as all the models tested had the same area and were tested in the same flow field.

SYMBOLS

M	Mach number
α	angle of attack relative to geometric root chord line
C_L	lift coefficient
c_l	section lift coefficient
C_m	pitching-moment coefficient relative to $\frac{1}{4} \bar{c}$
C_D	drag coefficient
L/D	lift-drag ratio
S	model area
b	model span
c	local wing chord
c_{av}	average wing chord, S/b
$\bar{c}$	mean aerodynamic chord
A	aspect ratio simulated by wing model
Λ	sweepback angle of $\frac{1}{4}$ - chord line
λ	taper ratio
η	nondimensional spanwise coordinate

Subscripts:

A	airplane
c/4	quarter-chord line
i	ideal
MAX	maximum
o	zero lift
r	root
t	tip

WING DESIGN PROCEDURE

Arbitrary design features.- Current wing design and construction trends influenced the choice of most of the physical characteristics of the models in order that the investigation might have more practical value. On this basis the following wing design parameters were selected:

Quarter-chord sweep angle, deg	45
Taper ratio	0.25
Aspect ratios	4, 6, and 8
Airfoil section	NACA 65A-series, $a = 0.5$ mean camber line
Thickness ratio	8 percent chord for $A = 6$

Determination of twist and camber.- If a valid comparison is to be made between wings of any series it is necessary that the wings be designed to the same objective. For present purposes it was assumed that the three cambered and twisted wings of different aspect ratio were all to produce a lift coefficient of 0.3 at a Mach number of 0.90.¹ Wing theory indicates that any wing operates with least drag due to lift at a given lift coefficient when its span load distribution is elliptical at that lift coefficient. Therefore the three wings were each designed

¹Actually the design Mach number was found to be of secondary importance so long as it is below the wing force-break Mach number since calculations of the amount of twist required to produce the same lift distribution at different subsonic Mach numbers showed that the amount of twist was relatively insensitive to Mach number for the wing planforms considered herein.

to produce very nearly an elliptical span load distribution at a lift coefficient of 0.3 at a Mach number of 0.90. Computations were carried out by the Weissenger method and the three-dimensional Prandtl-Glauert compressibility rule (ref. 1) to determine the twist required to produce elliptical loading at the design conditions. As might be expected these computations showed that a nonlinear variation of twist angle with span would be required. However, further calculations indicated that a very close approach to elliptical loading could be obtained by employing linear twist of the correct amount; in the interests of reducing model construction difficulties, therefore, the models were built with linear twist.²

Results of the twist calculations are shown in terms of the predicted span load distributions at design conditions for the three cambered and twisted wings in figures 1, 2, and 3. For purposes of reference these three twisted wings are referred to as wings I, II, and III throughout this paper corresponding to the models of aspect ratio 4, 6, and 8, respectively. The plane wing of aspect ratio 6, corresponding to wing II in all respects except that it had no twist nor camber is referred to as wing IV. The amount of linear twist used in wings I, II, and III was determined as follows: First, the additional loadings determined by the plan-form shapes were plotted against wing semispan together with the elliptical load distribution corresponding to the same lift coefficient (refer to figs. 1, 2, and 3). Next, the difference between the additional and the elliptical loading curves was plotted and the total area between this curve and the X-axis, disregarding sign, was determined. Finally the degree of linear twist was determined as that twist for which the total area between the basic loading curve and the X-axis, disregarding sign, was equal to the foregoing area. The total loading curves were then obtained by simply adding the additional and the basic loading curves. Note in the figures that the calculated total loading curves of wings I, II, and III are very nearly elliptical. This result is due largely to the fact that very little linear washout (never more than 2.3°) was required to obtain approximately elliptical loading for the particular combination of sweepback and taper ratio chosen for these wings. Mention might also be made that the total loading for wing IV at design lift also is shown in figure 2.

Besides providing for the attainment of an elliptical load distribution, it was desired to use camber in an effort to keep the wings from exhibiting an early leading-edge stall or an early loss in longitudinal

² As used herein twist refers to the relative inclination of the section zero-lift lines along the span measured in planes parallel to the stream direction.

stability due to tip stalling. The amount of camber used was determined as follows: From the total loading curves of wings I, II, and III (figs. 1, 2, and 3) the variation of section lift coefficient at design conditions was determined and plotted against semispan as is shown in figure 4. The fact that all three wings have almost identical calculated variations of section lift coefficient with span is not surprising inasmuch as all the wings had very nearly elliptical total loadings and exactly the same taper ratio. Because these section loading curves were the same, the same amount of camber was chosen to be built into wings I, II, and III. This amount of camber is shown by the dashed straight line in figure 4. As may be noted, enough camber was given to the sections on the inboard 50 percent of the semispan so that these sections would operate at their ideal angles of attack when the wings were at design lift coefficient. On the outboard 50 percent of the semispan the sections were arbitrarily given sufficient camber to produce an over-all linear variation of camber with span. This was done primarily for reasons of ease of construction of the models. Actually the flow conditions near the tip of a physical sweptback wing appear to be extremely complicated so as to prohibit a rigorous computation of the camber required. At the tip, of course, a physical wing will not produce lift regardless of section shape. However, general knowledge of the flow on swept wings indicates clearly that the tip sections usually stall first and therefore it appears reasonable to supply them with a generous amount of camber. Based on the foregoing, wings I, II, and III were all built with a linear variation of camber with span for which the root and tip section design lift coefficients due to camber were 0.24 and 0.44, respectively. Because of the variation in camber with span, the geometric washout became different from the aerodynamic washout and this was accounted for in the design of the models by applying the computed linear aerodynamic twist to the spanwise variation in the attitude of the local section zero-lift lines.

Determination of thickness ratios.--Elementary considerations show immediately that for a series of wings of equal area, of equal weight, of similar construction, and carrying the same load, the thickness ratio must increase with increase in aspect ratio. This is readily seen by examining the root bending moments. A high-aspect-ratio wing has greater root bending moments because its centroid of lift is at a greater linear distance from the wing root. If the wing is a shell-like structure of given total weight and the maximum root stress is fixed, this means that the total linear thickness of the wing root section must be greater for a high-aspect-ratio wing; this, coupled with the shorter linear length of the root chord, leads to considerably greater thickness ratios for the root sections of high-aspect-ratio wings than for the root sections of low-aspect-ratio wings. Similar conclusions apply to wing stations other than at the root. An analysis based on somewhat more rigorous assumptions but still representing an approximation was actually used to determine the variation of thickness ratio with aspect ratio. This

analysis is described in reference 2. The procedure applied in the present instance is described in the next paragraph.

The wing of intermediate aspect ratio of 6 was assigned a thickness ratio of 10 percent of the chord perpendicular to the 40-percent-chord line; for the taper ratio (0.25) and sweepback (45°) selected, this thickness is equal to 7.76 percent of the streamwise chord. Actual wing thicknesses on present-day swept-wing airplanes were used as a guide in making this selection. From this point two different criteria were used to determine the thickness ratios of the aspect-ratio-4 wing and the aspect-ratio-8 wing. In the first criterion, all the wings were assumed to have equal structural weight and to experience an equal forward movement of the aerodynamic center (measured in dimensional movement of the neutral point, not as a fraction of $\bar{c}$) due to aeroelastic distortion. In the second criterion, all the wings were assumed to have equal structural weight and to incur equal maximum stress at the wing root section per unit g increment in normal acceleration. The results of these calculations are shown in table I. As indicated therein the computed thickness ratios were rounded off so that wings I, II, and III were built to thickness ratios of 4, 8, and 12 percent of the chord measured in the streamwise direction.

APPARATUS

An F-51D airplane was used as the wing-flow vehicle. The four models were mounted successively on a specially prepared test panel on the right wing well outboard of the propeller wake. A photograph of one of the models mounted on the F-51D airplane is given in figure 5.

The lift, chord force, and pitching moment acting on the models were measured by a strain-gage balance and recorded by a six-channel Heiland Oscillograph Recorder. A slide-wire potentiometer, also recording on the Heiland Oscillograph Recorder, was used to measure the position of the models relative to the airplane X-axis and the angle of flow at the model location was determined by a calibrated, rectangular, freely floating vane located $22\frac{1}{2}$ inches outboard of the model location (fig. 5). Standard NACA recording instruments were employed to measure the airspeed, altitude, three components of acceleration, and free-air temperature experienced by the airplane during the test runs. Continuous records of all the foregoing quantities were obtained as the models were oscillated continuously at approximately constant angular velocity through a large angle-of-attack range and as the airplane was flown through approximately its permissible speed range. All the records were synchronized by a common $\frac{1}{10}$ -second timer and, in addition, the Heiland galvanometer traces were synchronized by a separate $\frac{1}{100}$ -second timer.

Detailed geometric characteristics of the four models tested are listed in table II. The plan forms and root and tip section shapes of the models are shown in figure 6. Two photographs of the four models are included in figure 7. The models were machined from solid stainless steel and a thin (0.034 - in. thick) circular end plate with its bottom surface chamfered to a sharp edge was attached rigidly to the base of each model. The diameter of this end plate was equal to the root chord of model I and the same end plate was used with all the models in order to keep the tare drag nearly constant. The bottom of the end plate was about $\frac{1}{16}$ - inch above the F-51D wing surface. A cylindrical mounting boss 1 inch in diameter spanned this $\frac{1}{16}$ - inch gap and connected the model-end plate combination to the strain-gage balance located in an insulated box inside the F-51D wing. The temperature inside the box was held nearly constant by a thermostatically controlled electrical heating system in order to minimize strain-gage zero shifts due to temperature changes.

TESTS

Data were recorded during dives and pull-outs of the F-51D airplane covering an altitude range of about 28,000 feet to 18,000 feet and a speed range from about 220 miles per hour calibrated airspeed to a true airplane Mach number of 0.75. During these runs the average Mach number on the models varied from about 0.65 to 1.17. The variation of local Mach number just outside the wing boundary layer at the spanwise model station is shown as a function of chordwise position at several airplane Mach number and lift conditions in figure 8. The drop-off in flow velocity with increasing vertical distance above the test panel is shown by figure 9. It may be noted that inasmuch as the local Mach number increases with rearward movement along the chord but decreases with increasing vertical distance above the wing surface, 45° sweptback wing models experienced very little change in local Mach number along their spans at any constant airplane condition. Typical variations of Reynolds number with Mach number for the four models tested are shown in figure 10. The model angle-of-attack range was from about -8° to 32°. Measurements were made with the models in both the smooth and rough surface condition; for the rough condition, carborundum particles having maximum dimensions between 0.003 in. and 0.005 in. were scattered uniformly on thinned shellac over the first 5 percent of the model chords on both upper and lower surfaces.

ACCURACY AND DESCRIPTION OF MEASUREMENTS

The major variables were estimated to have been measured with the following accuracy:

	<u>M = 0.65</u>	<u>M = 1.17</u>
Mach number	± 0.01	± 0.01
Angle of attack, deg	± 0.3	± 0.3
Lift coefficient	± 0.04	± 0.01
Pitching-moment coefficient	± 0.04	± 0.01
Chord-force coefficient	± 0.006	± 0.002

These values refer to absolute accuracy; relative accuracy between data points at a given Mach number should be much better because of the absence of zero shifts that occurred over relatively long periods of time. Also it is to be noted that the accuracy of the drag-coefficient data presented herein is slightly less than that of the chord-force coefficient quoted in the preceding table inasmuch as the drag coefficients had to be obtained from a long calculation procedure in which many variables were involved including the lift, chord force, angle of attack, and airplane acceleration. However the zero-lift drag coefficients were determined quite accurately since, in general, only the chord force and airplane longitudinal acceleration were involved in their determination.

Other limitations applying to the measurements are the following: The pitching moments were actually measured about axes approximately 25, 55, and 85 percent mean aerodynamic chord ahead of the leading edge of the mean aerodynamic chords of models I, II, and III, respectively, because of the model-balance configuration used which, in turn, was dictated largely by the F-51D wing external and internal geometry. Consequently, it was necessary to use insensitive settings of the pitching-moment balance in order to measure the large pitching moments and this led to a reduction in accuracy of the pitching-moment data. The pitching-moment data have been transferred to the standard $\frac{1}{4}$ -chord mean aerodynamic chord reference axis and are presented so herein. No tare forces were measured or subtracted from the total forces of the model-end-plate combination. As mentioned previously, however, the same end-plate was used with all the models so that the main effect should be that all the drags presented are too high by nearly the same amount. No corrections were applied for the effects of model flexibility. This procedure is felt to be justified in view of the model construction (solid steel) and in view of the altitude range of the tests (28,000 feet to 18,000 feet).

The angles of attack presented herein are given with respect to the geometric root chord line of each wing model.

PRESENTATION AND DISCUSSION OF RESULTS

Lift characteristics.- The basic aerodynamic data for the smooth surface condition, including the variation of lift with angle of attack, are given in figure 11. Lift-curve slopes measured from $C_L = 0$ to $C_L = 0.3$ are plotted together on figure 12 and these same curves are compared singly with subsonic theory in figure 13. Figure 14 gives the maximum lift coefficients measured during the tests plotted against Mach number. Above a Mach number of 1.0 the lift was still rising at the highest test angle of attack of 32° so that $C_{L_{max}}$ was not determined at supersonic speeds; however, the values of C_L at $\alpha = 32^\circ$ at these speeds are probably not far below $C_{L_{max}}$. The variations in angle of attack for zero lift with Mach number are shown in figure 15.

In general the lowest-aspect-ratio, thinnest wing (wing I) showed the best over-all lift characteristics. The maximum lift was highest, particularly in the transonic speed range; the variations of lift with angle of attack and of lift-curve slope with Mach number were smoothest; and the variation in angle of zero lift with Mach number was smaller than for the other two cambered and twisted wings. The highest-aspect-ratio, thickest wing (wing III) showed the poorest lift characteristics and this was particularly true at the higher test speeds. A comparison of the data for the cambered and twisted wing of aspect ratio 6 (wing II) with that of the symmetrical wing of aspect ratio 6 (wing IV) indicates that adding camber and twist improved the maximum lift throughout the speed range and increased the lift-curve slope at supersonic speeds but decreased the lift-curve slope at subsonic speeds.

Drag characteristics.- The basic data of drag coefficient plotted against angle of attack are shown in figure 11. The variations of zero-lift drag coefficient with Mach number are given in figure 16 for the smooth wings. The effect of roughness on the zero-lift drag is shown in figure 17. The increment in drag coefficient measured from the zero-lift drag coefficient is plotted as a function of lift coefficient squared for Mach numbers of 0.70, 0.90, and 1.15 in figure 18. Also included on these figures are lines showing the increment of drag due to lift corresponding to no leading-edge suction (resultant force perpendicular to zero-lift chord plane) and that corresponding to the maximum leading-edge suction theoretically realizable. The latter curves were calculated from the relation $C_{D_i} = \frac{C_L^2}{\pi A}$ for the subsonic Mach numbers and from an adaptation of the theory of leading-edge suction on pointed wings (ref. 3) for the supersonic Mach number.

Wing IV, the symmetrical wing, exhibited the smallest drag at zero lift (fig. 16) at subsonic speeds. At supersonic speeds the thinnest wing (wing I) gave the lowest drag. These results indicate that the addition of camber and twist caused a large increase in the zero-lift drag and this can be seen directly by comparison of wing IV with wing II. This increase in drag might have been associated with some type of flow separation at zero lift on the cambered and twisted wings inasmuch as calculations of the induced drag corresponding to the computed basic lift distribution of wing II showed that this drag was negligible in comparison with the measured increment of drag due to camber and twist. It is suspected that the low Reynolds numbers of the tests contributed to the large zero-lift drag of the cambered and twisted wings; however, the trends of the data for wings II and IV at the lowest Mach numbers tested indicate Mach number may also have an appreciable influence on the drag due to camber and twist even at relatively low subsonic speeds. In this connection the use of a cambered section in a wing lowers the wing critical speed at zero lift (ref. 4). Of the cambered and twisted wings, wing I showed a transonic drag rise of the order of 50 percent as compared with about 200 percent for wing III. At supersonic speeds, wing III had over twice as much zero-lift drag as wing I. These results reemphasize the predominating influence of wing thickness ratio on drag at high speeds. The addition of roughness (fig. 17) generally caused a small increase in zero-lift drag. This effect was most pronounced for the symmetrical wing and lends weight to the suspicion that the smooth cambered and twisted wings might have been at a particular disadvantage in the present low Reynolds number tests.

With regard to drag due to lift (fig. 18), the data indicate that the use of camber and twist is an effective means of maintaining leading-edge suction. With increasing subsonic Mach number the wings lost their ability to produce leading-edge suction in the order of decreasing thickness ratio; that is, at $M = 0.90$ the 4-percent-thick wing (wing I) retained a large measure of leading-edge suction whereas the 12-percent-thick wing (wing III) had lost nearly all leading-edge suction. At $M = 1.15$ the cambered wings still showed greater leading-edge suction than the uncambered wing. At this Mach number both approximate theory and experiment predict lower drag due to lift for the low-aspect-ratio wing (I) than for the high-aspect-ratio wing (III).

Lift-drag characteristics.— Variations in lift-drag ratio with angle of attack are shown for Mach numbers of 0.70, 0.90, and 1.15 in figure 19. A comparison of the maximum lift-drag ratios of the three cambered and twisted wings is shown in figure 20. Figure 21 shows the effect of camber and twist on the maximum lift-drag ratio for the two wings of aspect ratio 6. Figure 22 is a plot of the lift coefficient for maximum lift-drag ratio against Mach number for the four models tested.

In general the maximum lift-drag ratios measured were appreciably lower at low speeds than would be expected for these wings alone. This is attributed to the low Reynolds numbers of the tests and to the presence of end-plate drag and possible interference drag in the results. The latter items, however, should have been nearly constant for all the models.

Above a Mach number of 0.70 the lowest-aspect-ratio, thinnest wing (I) was clearly superior to the two higher-aspect-ratio, thicker wings (fig. 20). The differences on a percentage basis were most marked at supersonic speeds although all the wings suffered very large losses in aerodynamic efficiency in passing through the high subsonic speed range. In figure 21 it can be seen that the maximum lift-drag ratio of the smooth symmetrical wing of aspect ratio 6 was greater than that of the cambered and twisted wing of equal aspect ratio and thickness ratio between Mach numbers of 0.71 and 1.02. This result can be traced to the greater zero-lift drag of the cambered and twisted wing which more than offset the lower drag due to lift of the cambered and twisted wing insofar as the determination of the maximum lift-drag ratio was concerned. The significance of the foregoing results is questionable, of course, in view of the low Reynolds numbers of the present tests. Below a Mach number of 0.71 the cambered and twisted wing was more efficient than the symmetrical wing as might be expected.

Pitching-moment characteristics.— The basic data of pitching-moment coefficient about the quarter-chord point of the mean aerodynamic chord plotted against angle of attack are shown in figure 11. A plot of the zero-lift pitching-moment coefficient as a function of Mach number is given in figure 23.

A study of figure 11 indicates that the lowest-aspect-ratio, thinnest wing had the most desirable pitching-moment characteristics throughout the angle-of-attack range at nearly every test speed. The stability tended to remain constant over a larger angle-of-attack range at low angles of attack and the changes in stability were less severe when the unstable break characteristic of high-aspect-ratio, highly sweptback wings occurred. The effect of adding camber and twist was generally beneficial at positive angles of attack (compare wing II with wing IV) inasmuch as the unstable break in the pitching-moment curves was thereby delayed to higher angles of attack and lift coefficients. It may be interesting to note that all the wings were stable at extreme angles of attack where wind-tunnel data are often not obtained. With regard to the pitching-moment trim changes with Mach number at zero lift (fig. 23), again wing I was superior to the other two thicker, higher-aspect-ratio, cambered and twisted wings (II and III). The data indicate that as the aspect ratio was increased (wings I, II, and III) the zero-lift pitching-moment coefficient changed from negative to positive. This trend suggests that the pitching moment due to camber predominates

at low aspect ratio whereas the pitching moment due to twist predominates at high aspect ratio.

CONCLUSIONS

A wing-flow investigation has been made to study the effects of aspect ratio of a 45° sweptback wing at transonic speeds for an assumed condition wherein the wing thickness ratio is allowed to vary in a logical manner with the aspect ratio. Three cambered and twisted, tapered, semispan wing models having aspect ratios of 4, 6, and 8 and respective constant thickness ratios of 4, 8, and 12 percent chord were tested. The investigation also included tests of a fourth model having a flat chord plane to determine the effects of camber and twist on the aerodynamic characteristics of a 45° sweptback wing of aspect ratio 6 and thickness ratio 8 percent chord. Inasmuch as the Reynolds numbers were low (230,000 to 620,000) the following conclusions must be considered preliminary until similar investigations are made at much higher Reynolds numbers:

(1) The lowest-aspect-ratio, thinnest, cambered and twisted wing tested had better aerodynamic characteristics than the other two higher-aspect-ratio, thicker wings in the following respects: (a) maximum lift, (b) variation of lift with angle of attack and Mach number, (c) variation in angle of attack for zero lift with Mach number, (d) drag at zero lift, (e) ability to develop leading-edge suction, (f) lift-drag ratio, (g) static longitudinal stability, and (h) trim change at zero lift with Mach number.

(2) Adding a small amount of camber and twist, (a) reduced the maximum lift-drag ratios at most speeds, (b) increased the zero-lift drag appreciably, (c) decreased the drag due to lift noticeably, (d) increased the maximum lift coefficient at all speeds, and (e) improved the pitching-moment variations with angle of attack or lift coefficient.

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TABLE I.- DETERMINATION OF WING THICKNESS RATIOS

[Based on constant total wing weight]

Aspect ratio	Criterion		Thickness ratios selected for models
	Constant forward shift in wing aerodynamic center due to flexibility	Constant maximum root stress per unit g normal acceleration	
4	0.0402	0.0430	0.04
6	^a .0776	^a .0776	.08
8	.1210	.1172	.12

^aAssumed.

TABLE II.- MODEL GEOMETRY

[End-plate diameter = 3.27 in.; end-plate
thickness = 0.034 in.]

Model	$\Delta c/4$, deg	λ	A	S, sq in.	b, in.	c_r , in.	c_t , in.	$\bar{c}$, in.	Streamwise NACA root section	Streamwise NACA tip section	Geometric washout, deg	Aerodynamic washout, deg	Design C_L at $M = 0.90$
I	45	0.25	4	8.33	4.08	3.27	0.82	2.29	65A(2.4)04, $a = 0.5$	65A(4.4)04, $a = 0.5$	3.3	2.0	0.3
II	45	0.25	6	8.33	5.00	2.67	0.67	1.87	65A(2.4)08, $a = 0.5$	65A(4.4)08, $a = 0.5$	3.6	2.3	0.3
III	45	0.25	8	8.33	5.77	2.31	0.58	1.62	65A(2.4)12, $a = 0.5$	65A(4.4)12, $a = 0.5$	3.6	2.3	0.3
IV	45	0.25	6	8.33	5.00	2.67	0.67	1.87	65A008	65A008	0	0	0



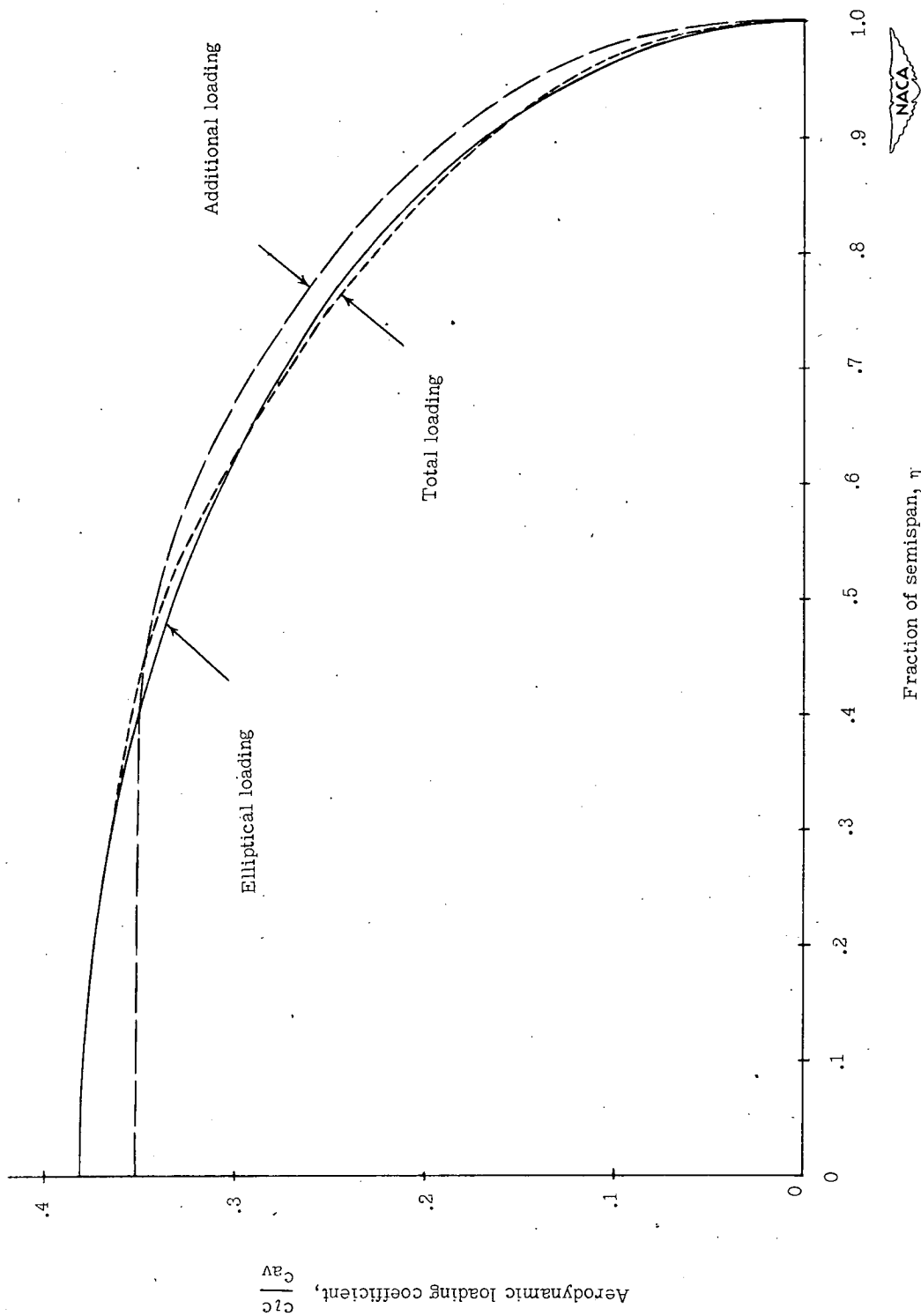


Figure 1.- Calculated aerodynamic loading of wing I at design condition ($C_L = 0.3$ at $M = 0.90$). Linear aerodynamic twist = 2.0° washout.

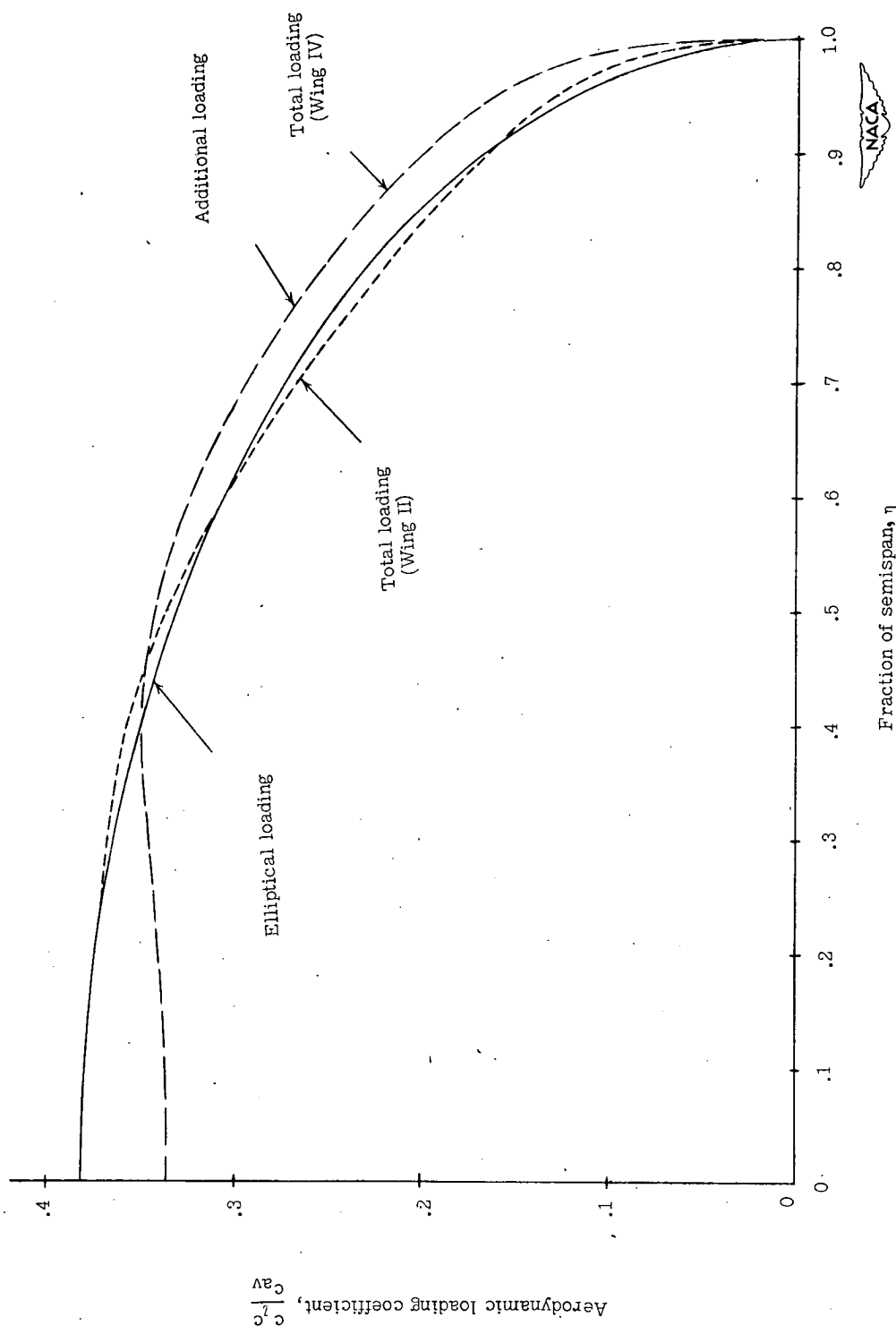


Figure 2.- Calculated aerodynamic loading of wing II at design condition ($C_L = 0.3$ at $M = 0.90$). Linear aerodynamic twist = 2.3° washout. Wing IV loading is also shown.

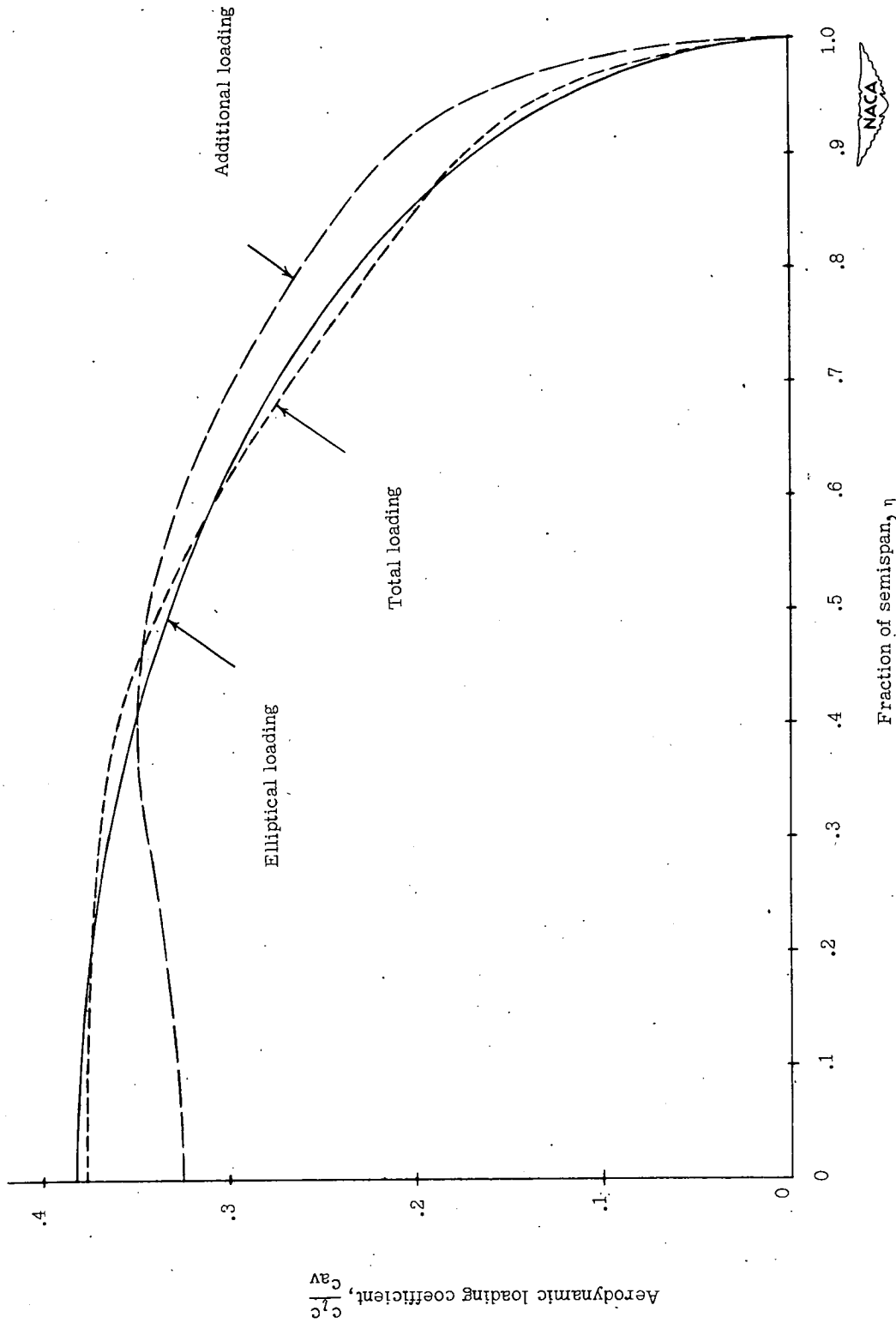


Figure 3.- Calculated aerodynamic loading of wing III at design condition ($C_L = 0.3$ at $M = 0.90$). Linear aerodynamic twist = 2.3° washout.

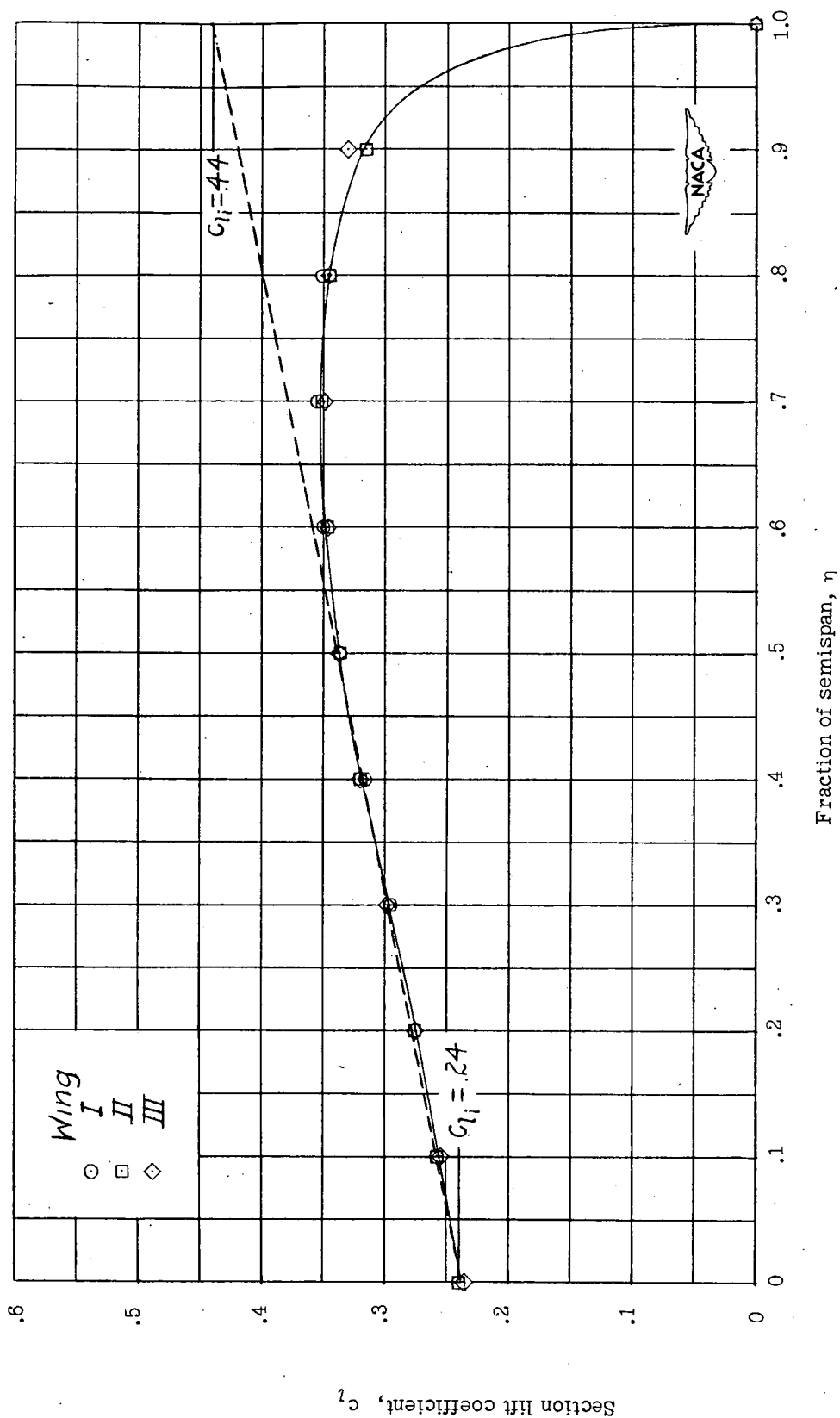


Figure 4.- Variation of calculated section lift coefficient with spanwise position for wings I, II, and III at design conditions ($C_L = 0.3$ at $M = 0.90$). Dashed line shows variation of design section lift coefficient due to camber selected for wing models I, II, and III.

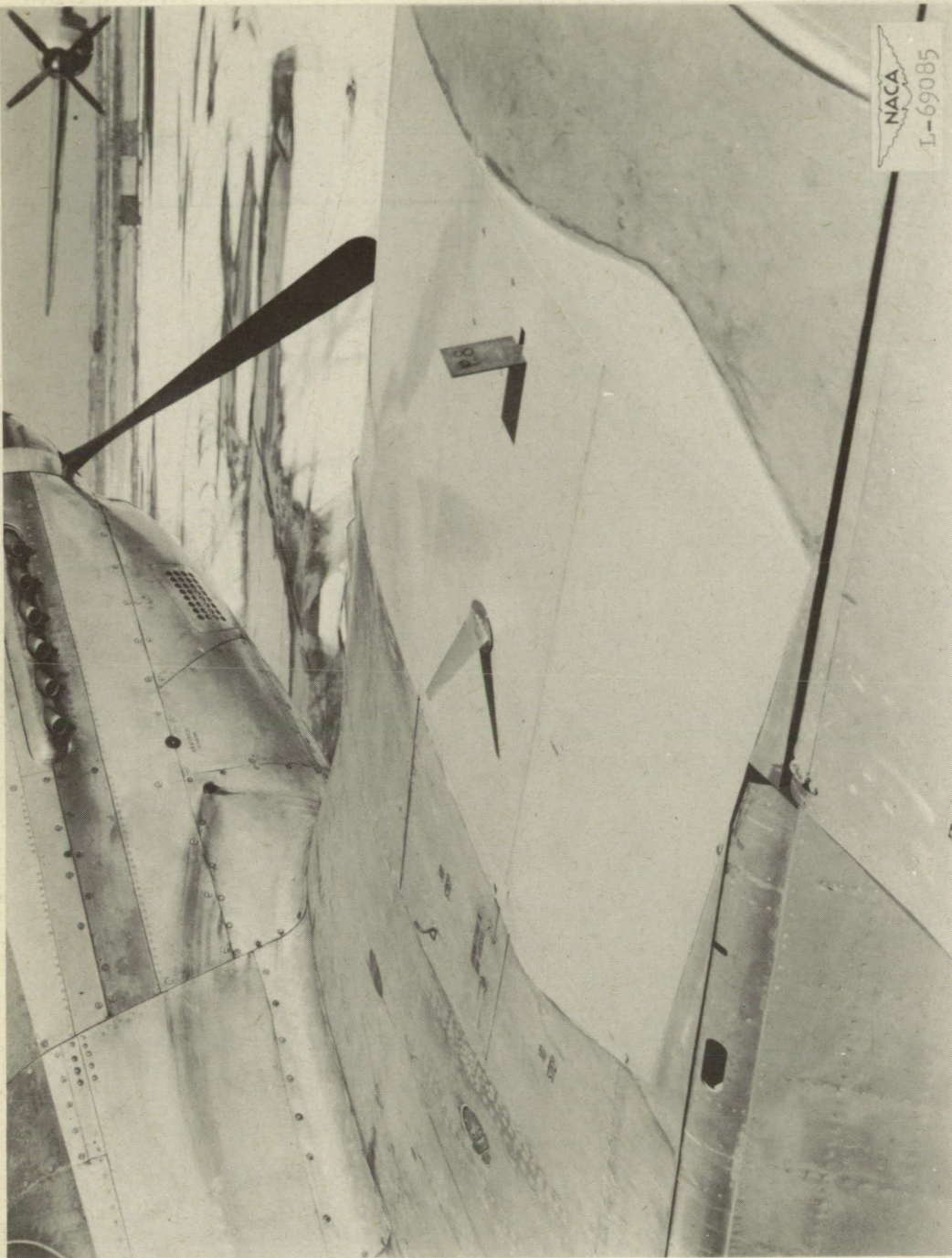


Figure 5.- View of model mounted on wing-flow test panel.

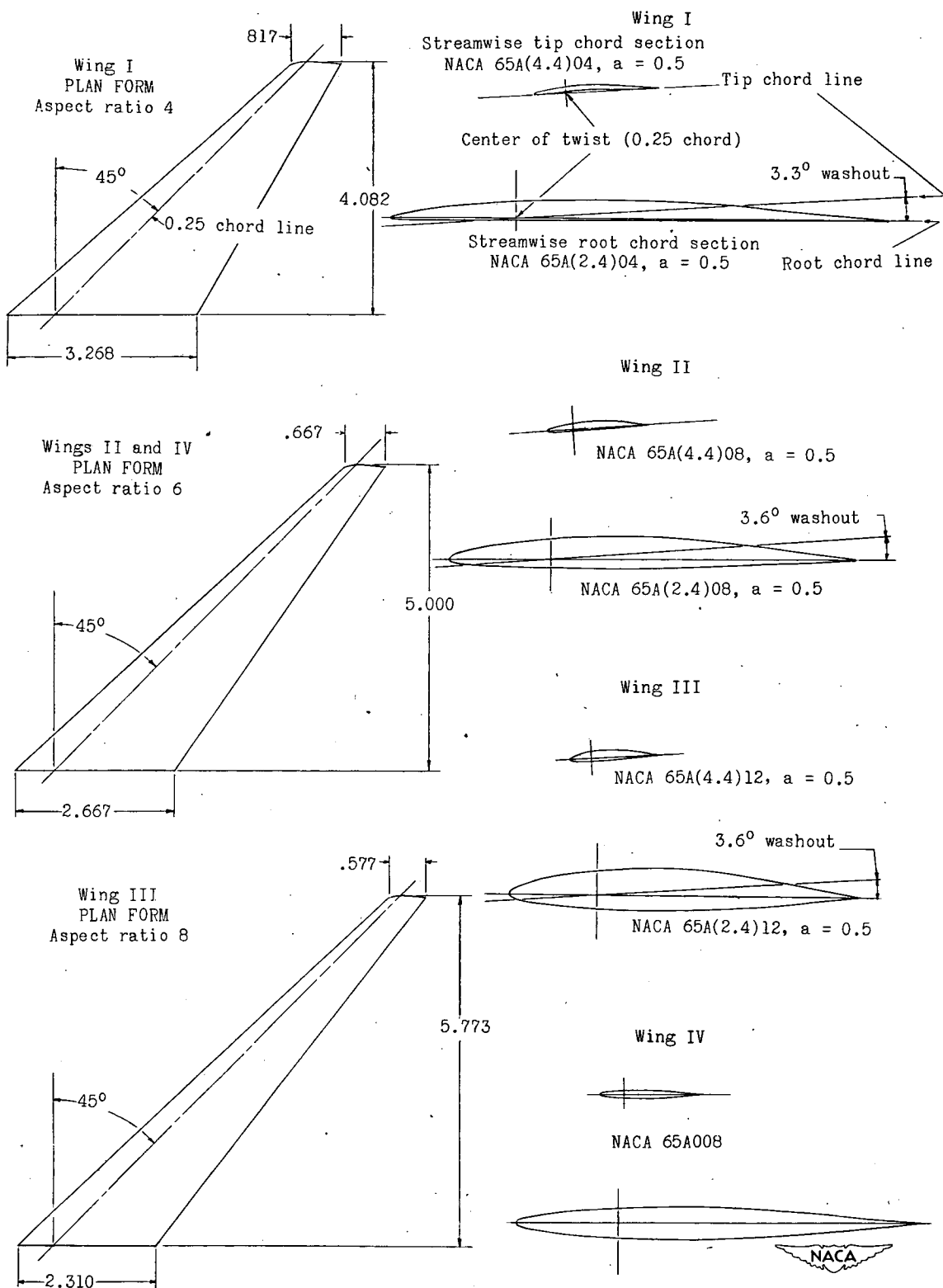
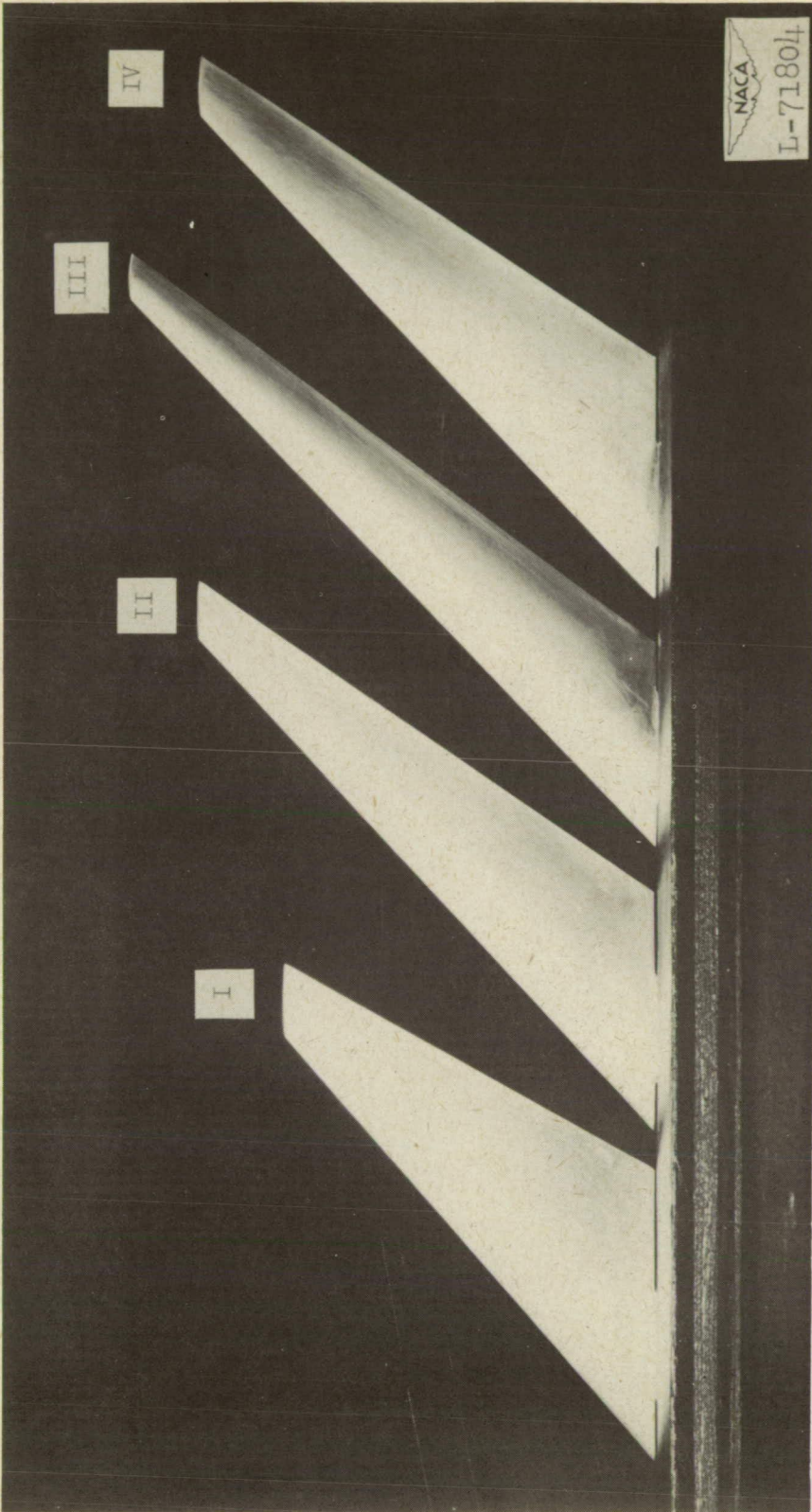
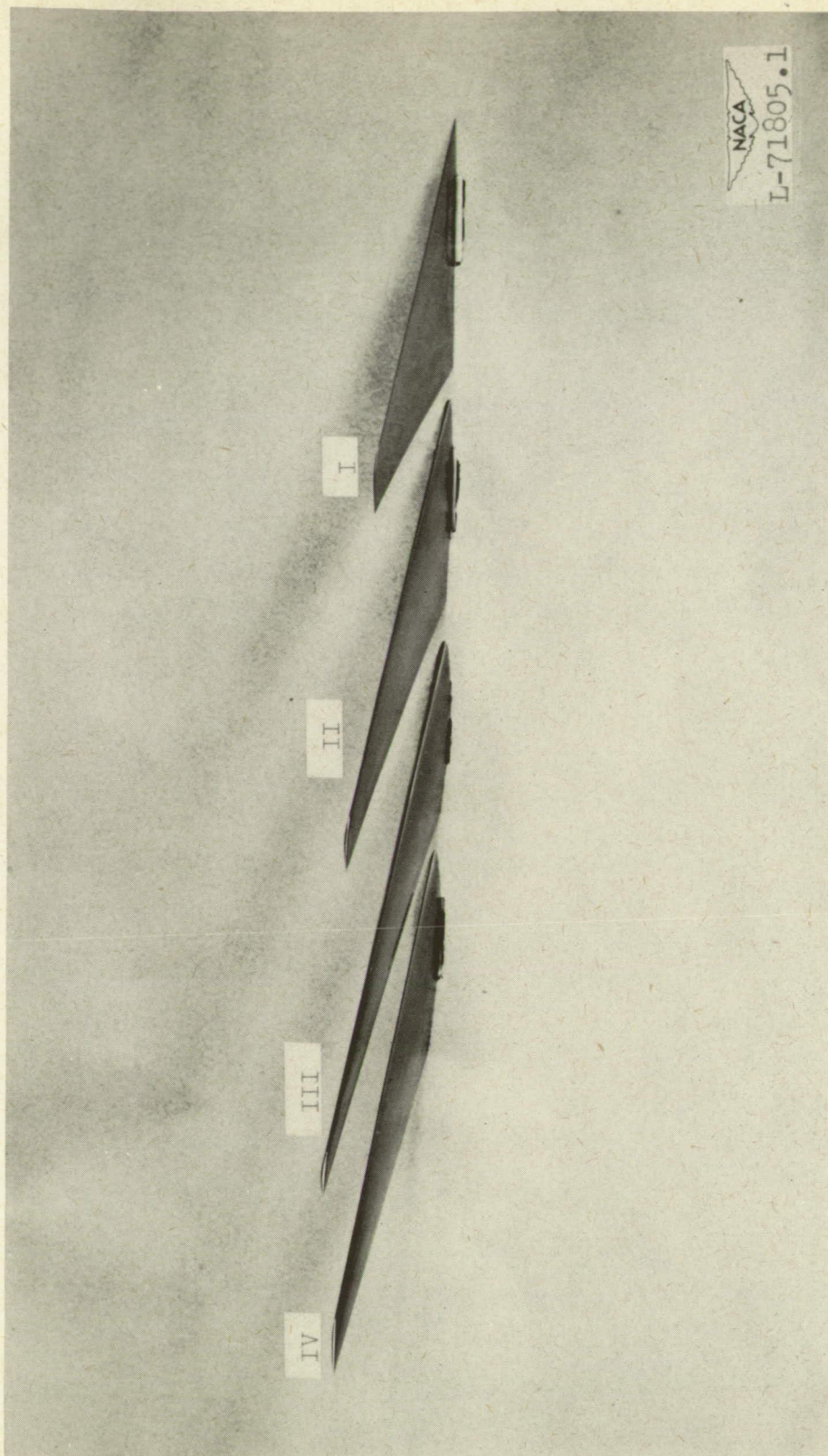


Figure 6.- Drawings of models tested.



(a) Plan-form view.

Figure 7.- Photographs of models tested.



(b) Tip view indicating relative thickness, camber, and twist.

Figure 7.- Concluded.

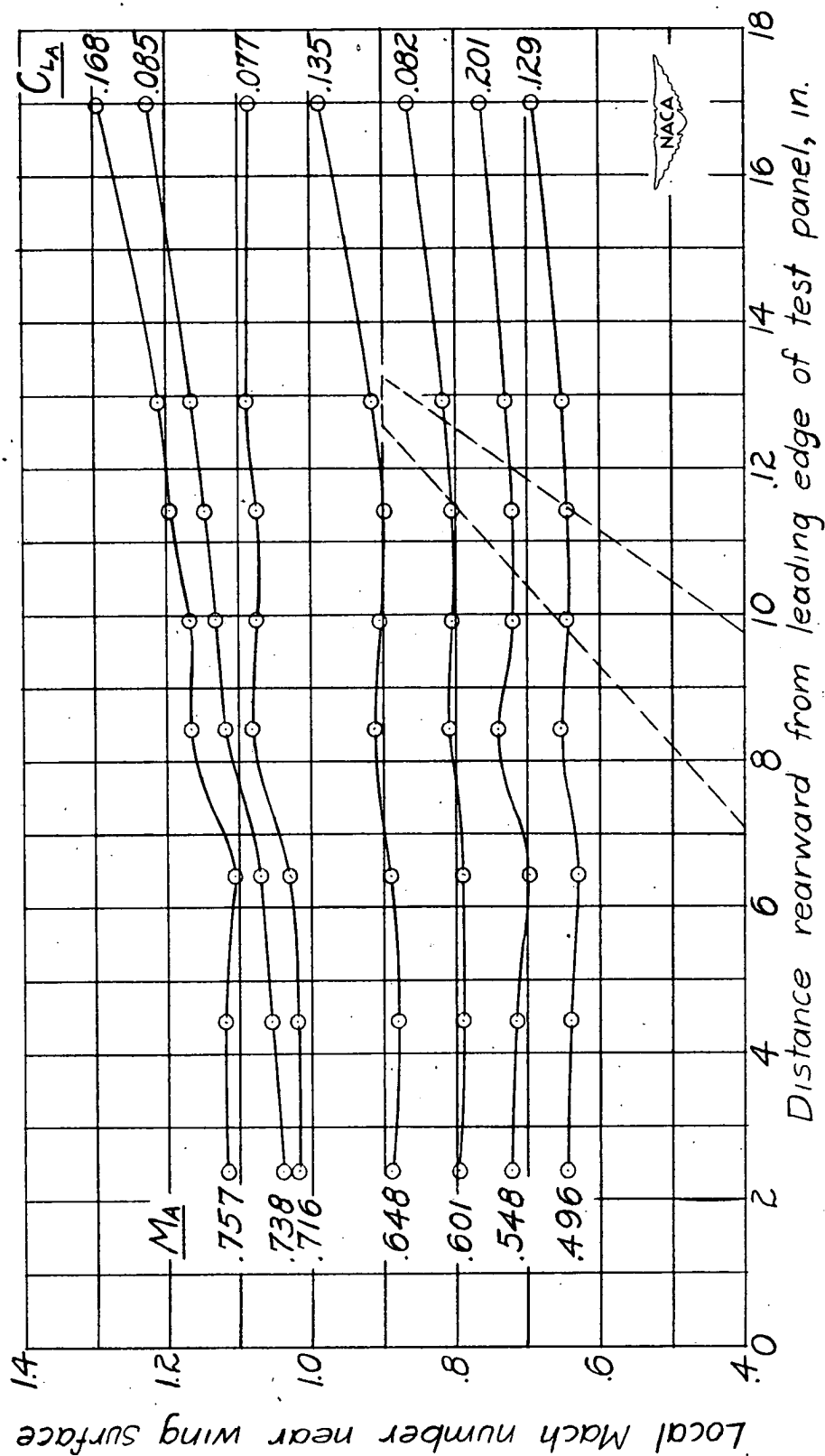


Figure 8.- Typical variations of local Mach number near wing surface with chordwise distance along wing surface for various airplane conditions as measured with model removed. Location of typical model indicated by sketch.

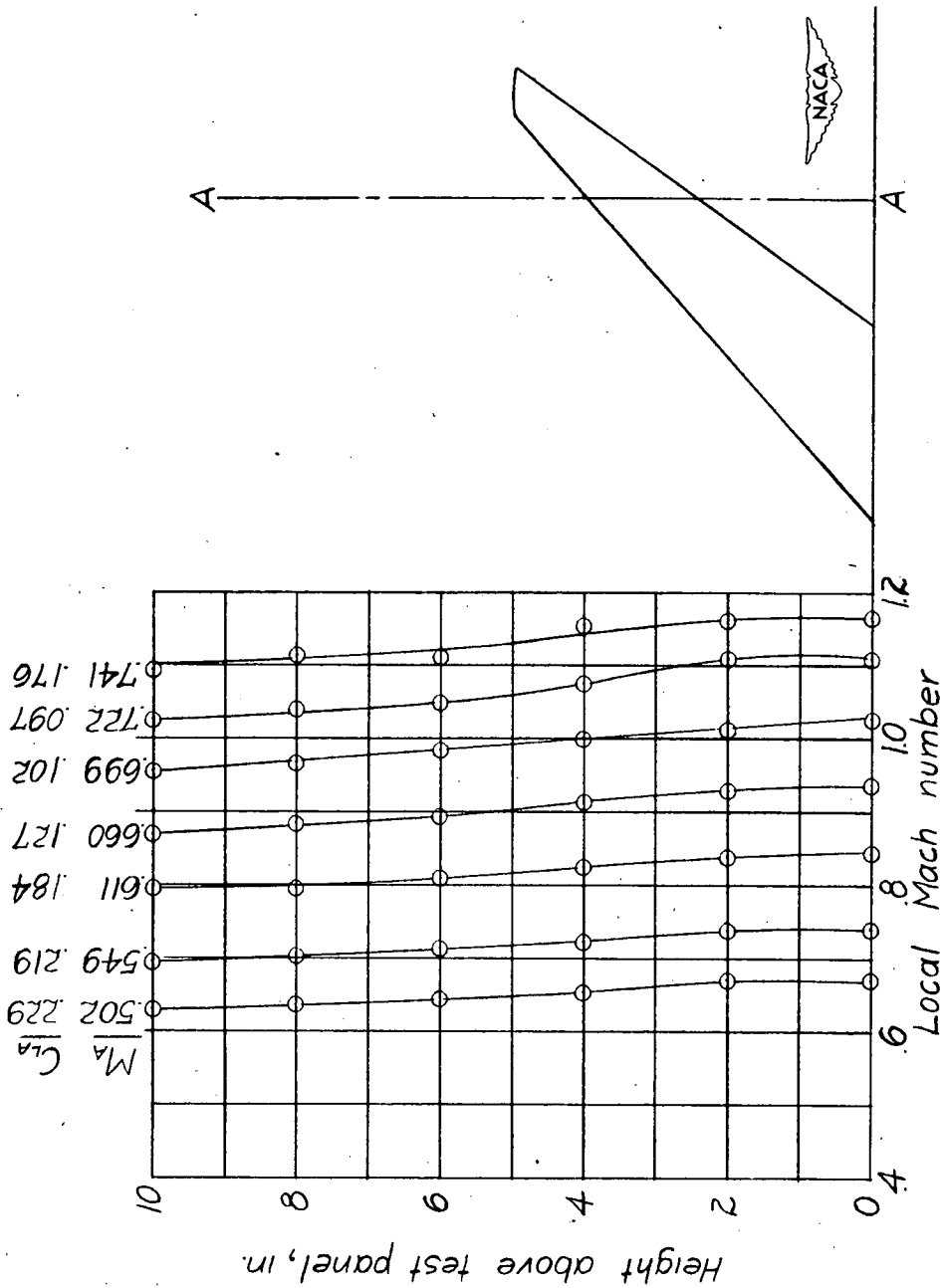


Figure 9.- Typical variations of local Mach number with vertical distance above test panel as measured at station AA with model removed. No allowance made for boundary layer over test panel which was approximately $\frac{1}{4}$ -inch thick.

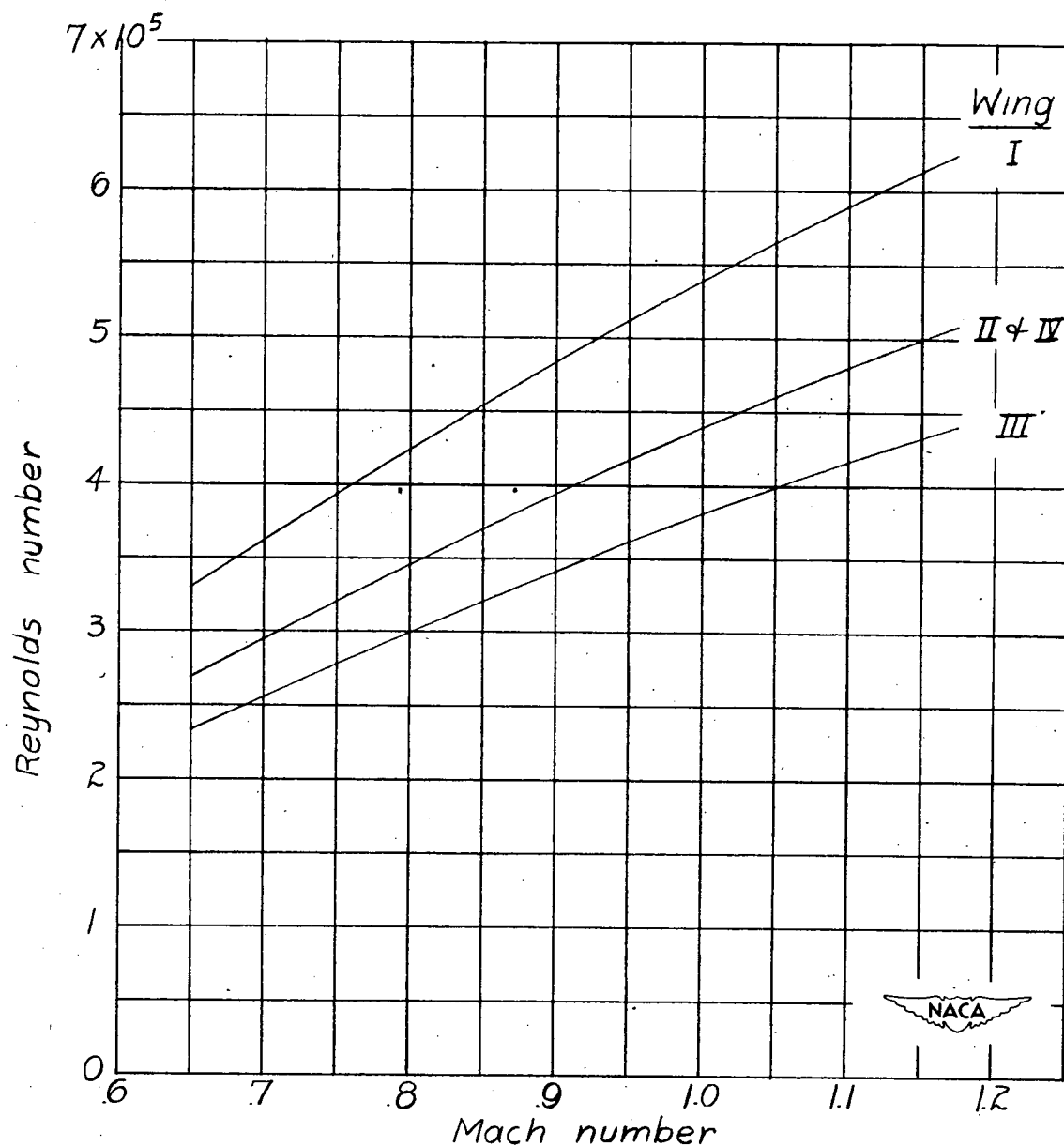


Figure 10.- Typical variation of Reynolds number with Mach number for wings tested.

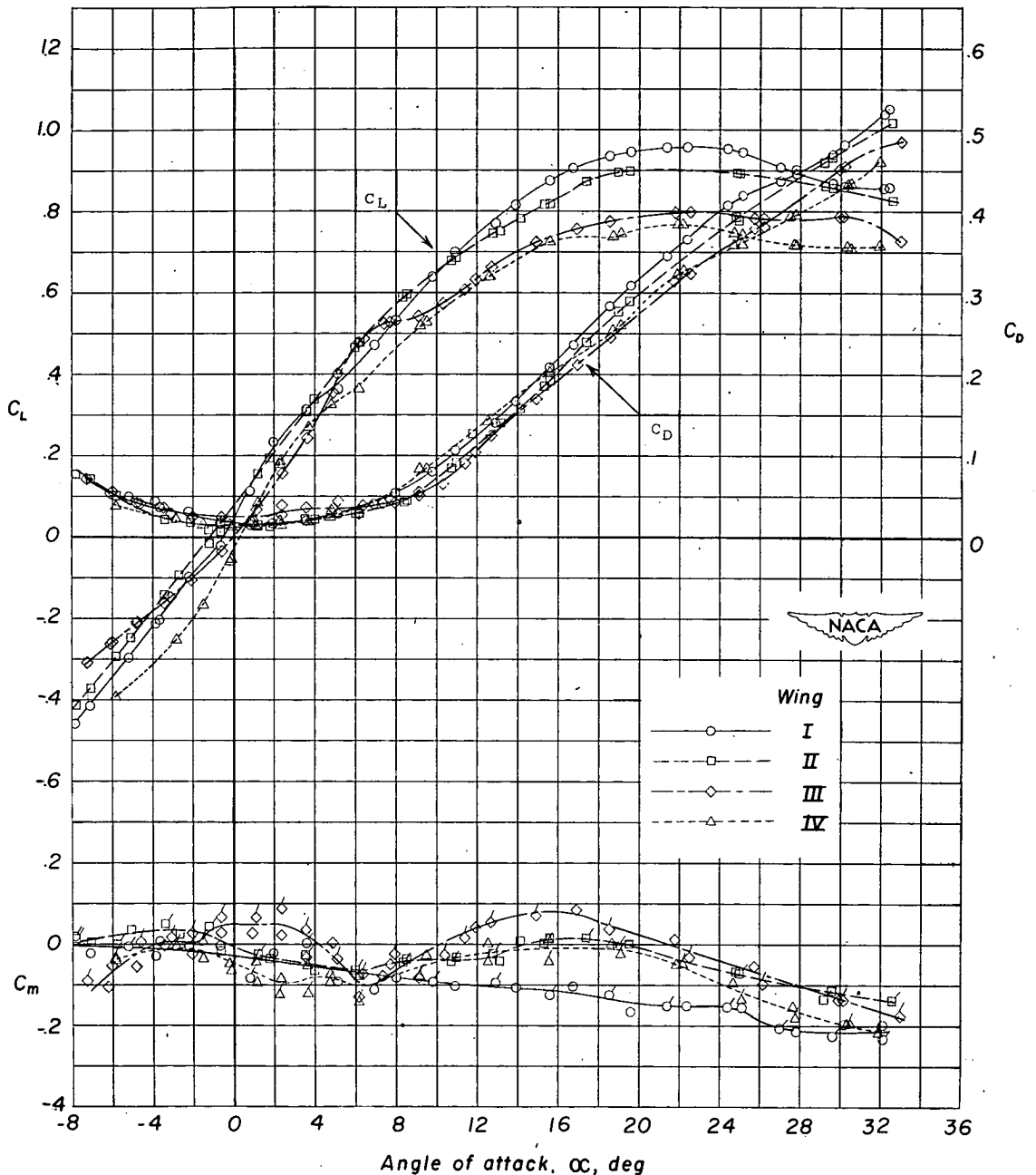
(a) $M = 0.65$.

Figure 11.- Variation of lift, drag, and pitching-moment coefficients with angle of attack throughout the test Mach number range for the series of models tested. On plot of C_m against α , plain and flagged symbols indicate increasing and decreasing angle of attack, respectively. Smooth surface condition.

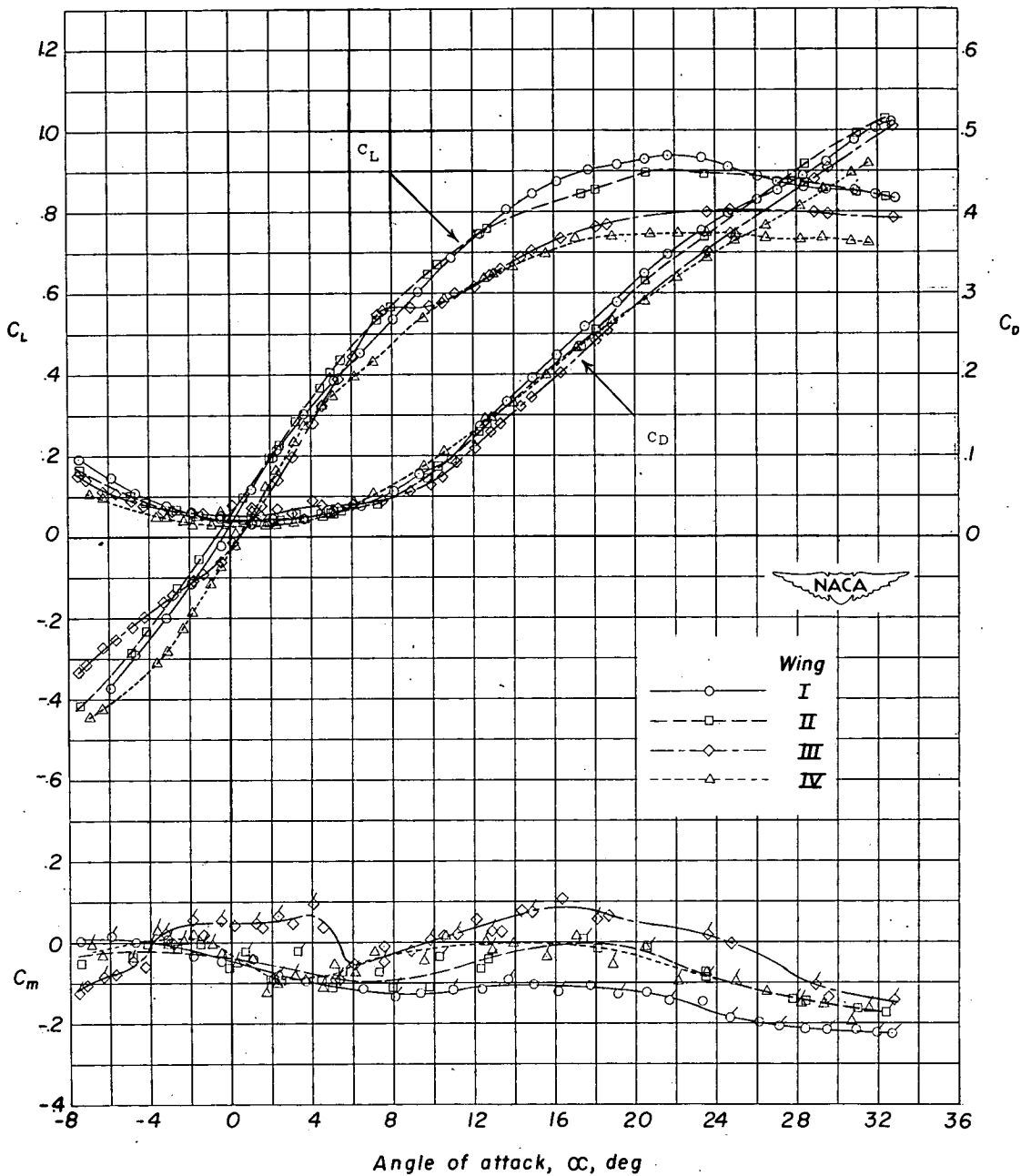
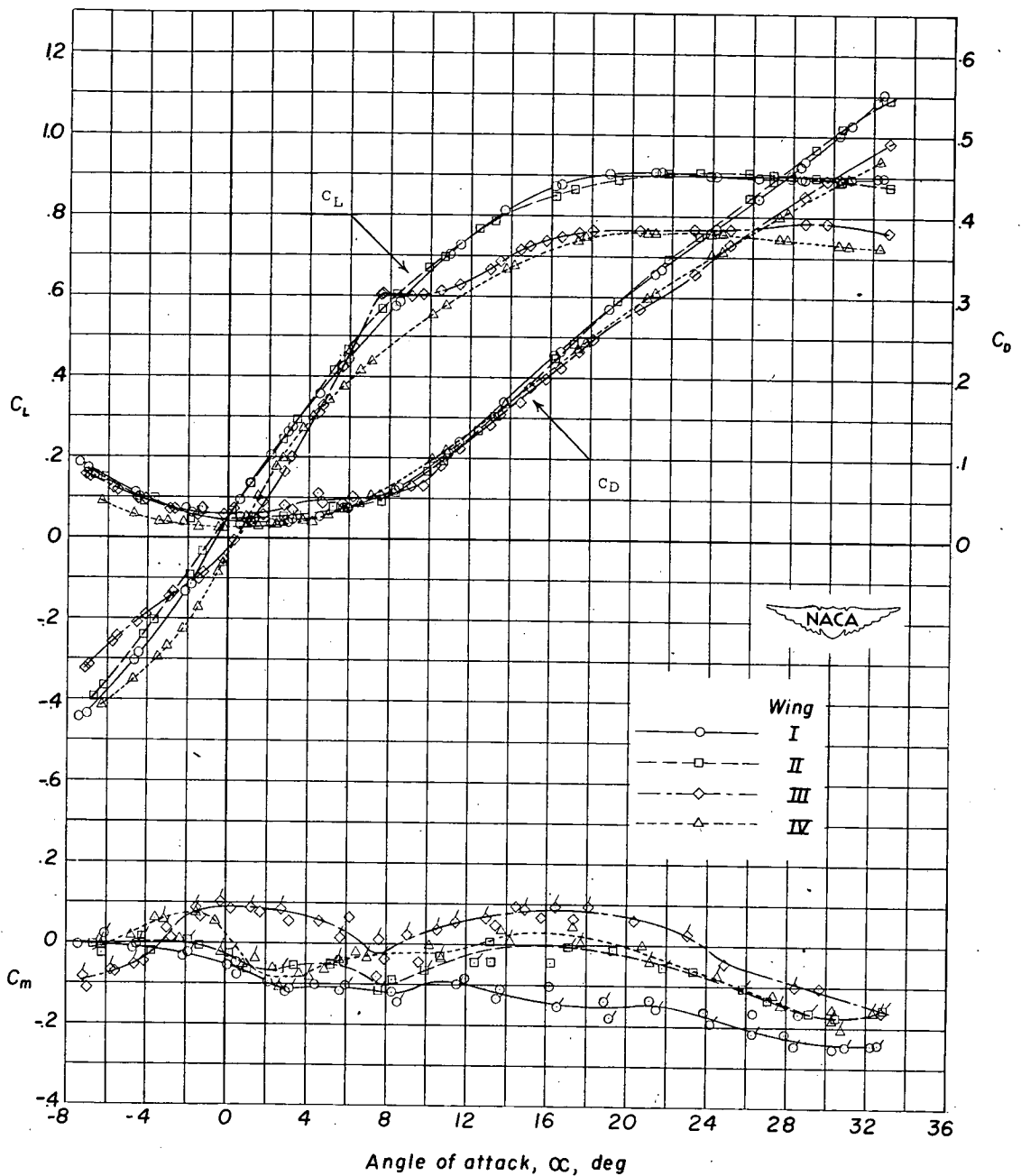
(b) $M = 0.70$.

Figure 11.- Continued.



(c) $M = 0.75$.

Figure 11.- Continued.

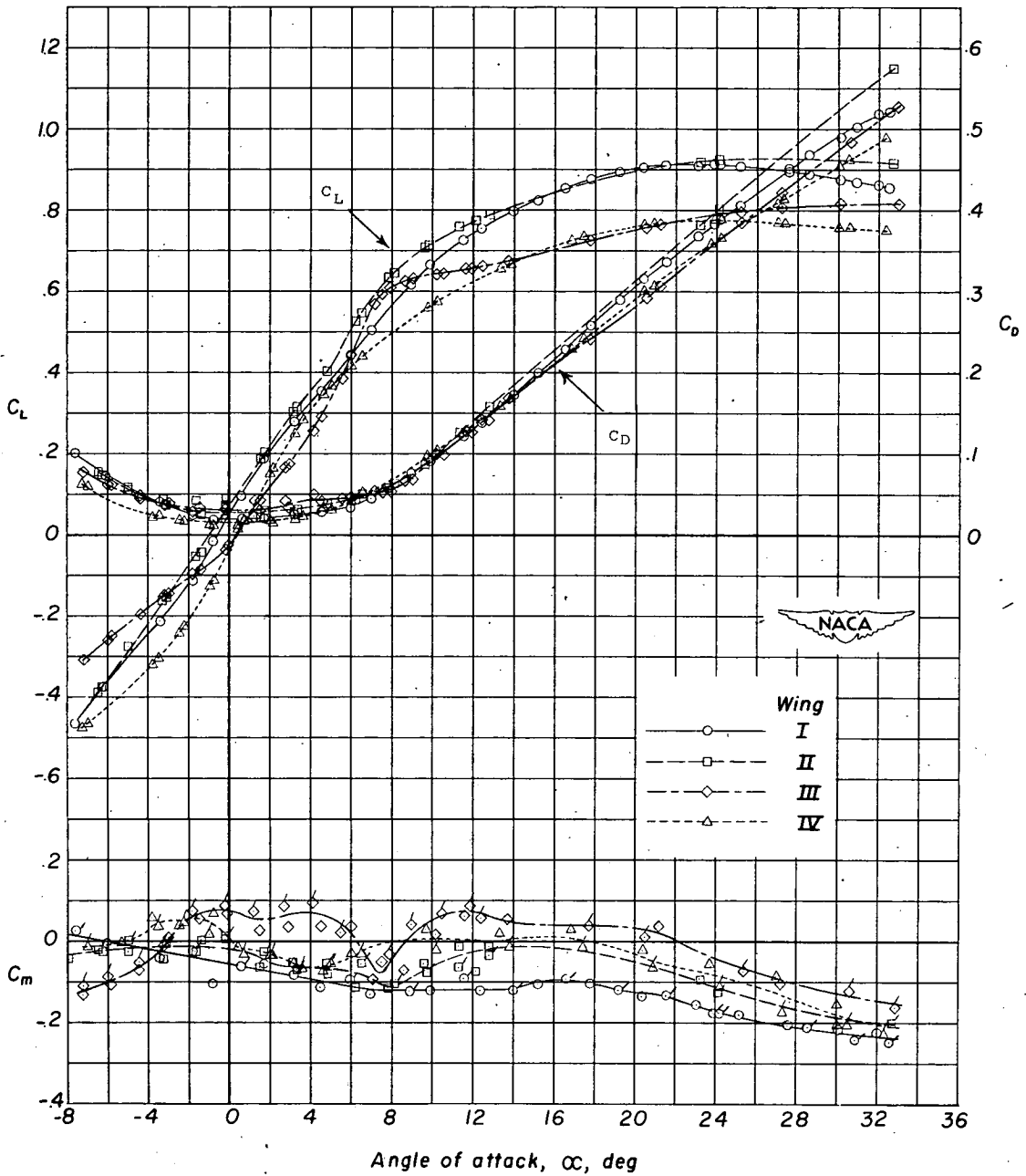
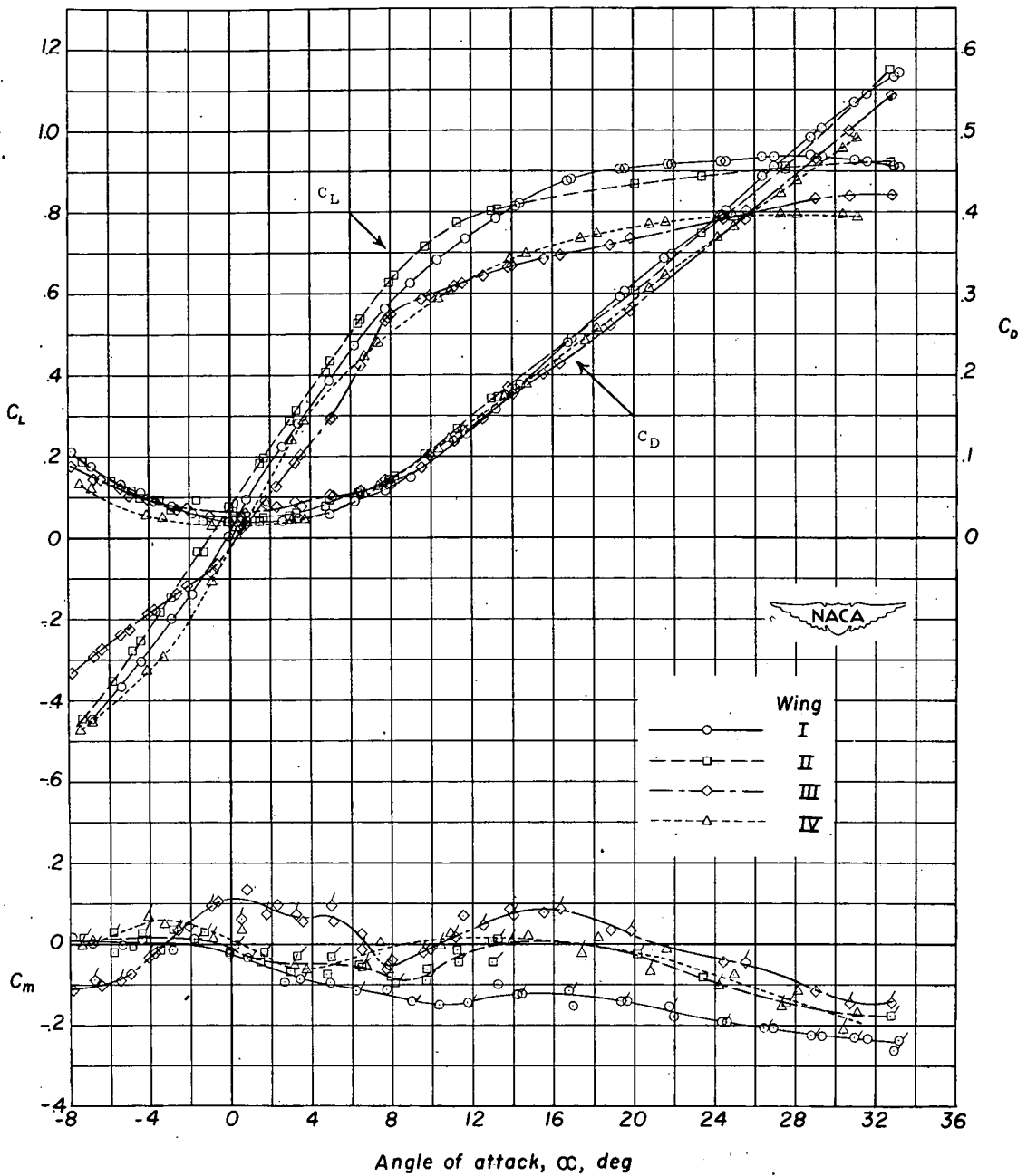
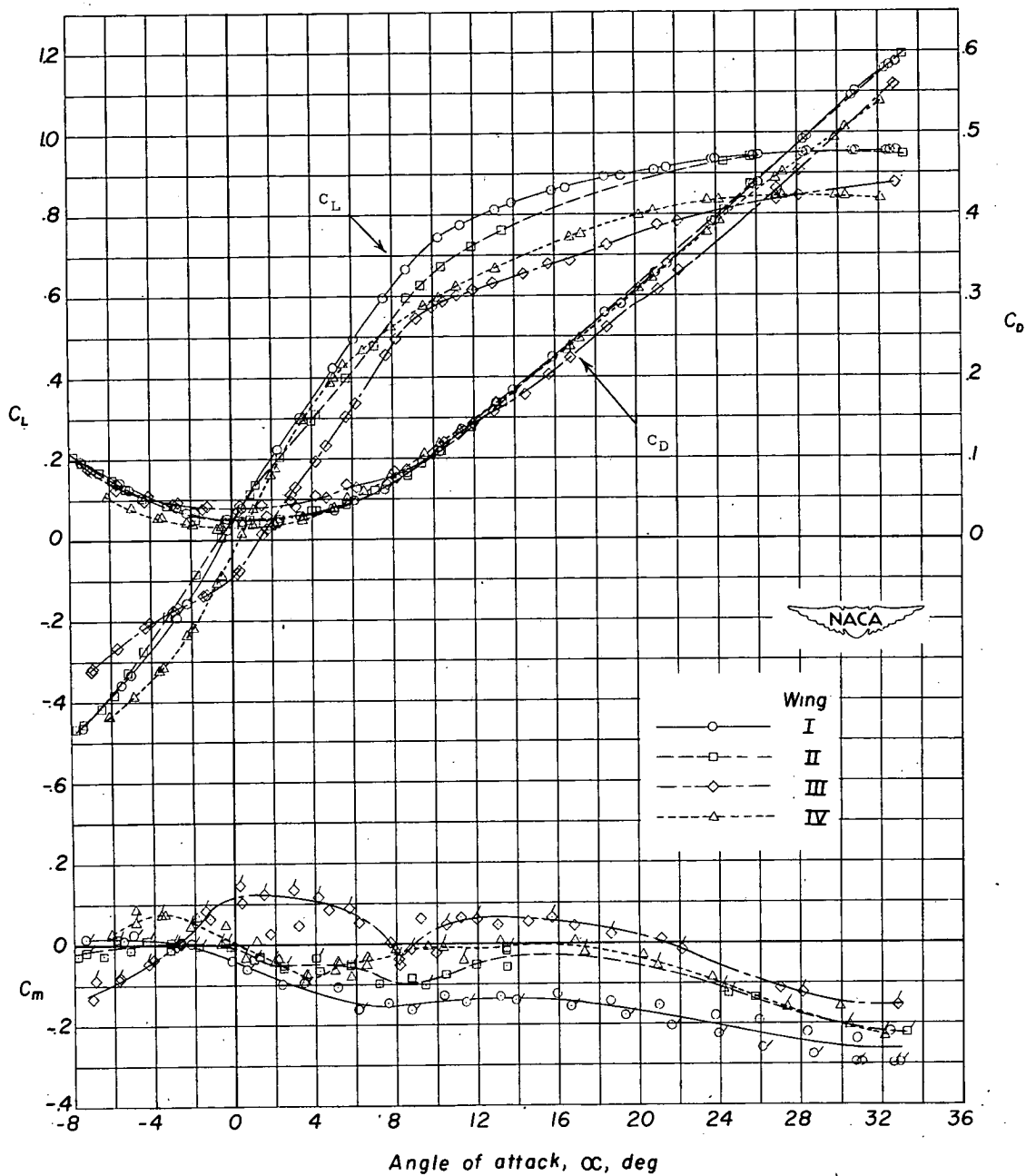
(d) $M = 0.80$.

Figure 11.- Continued.



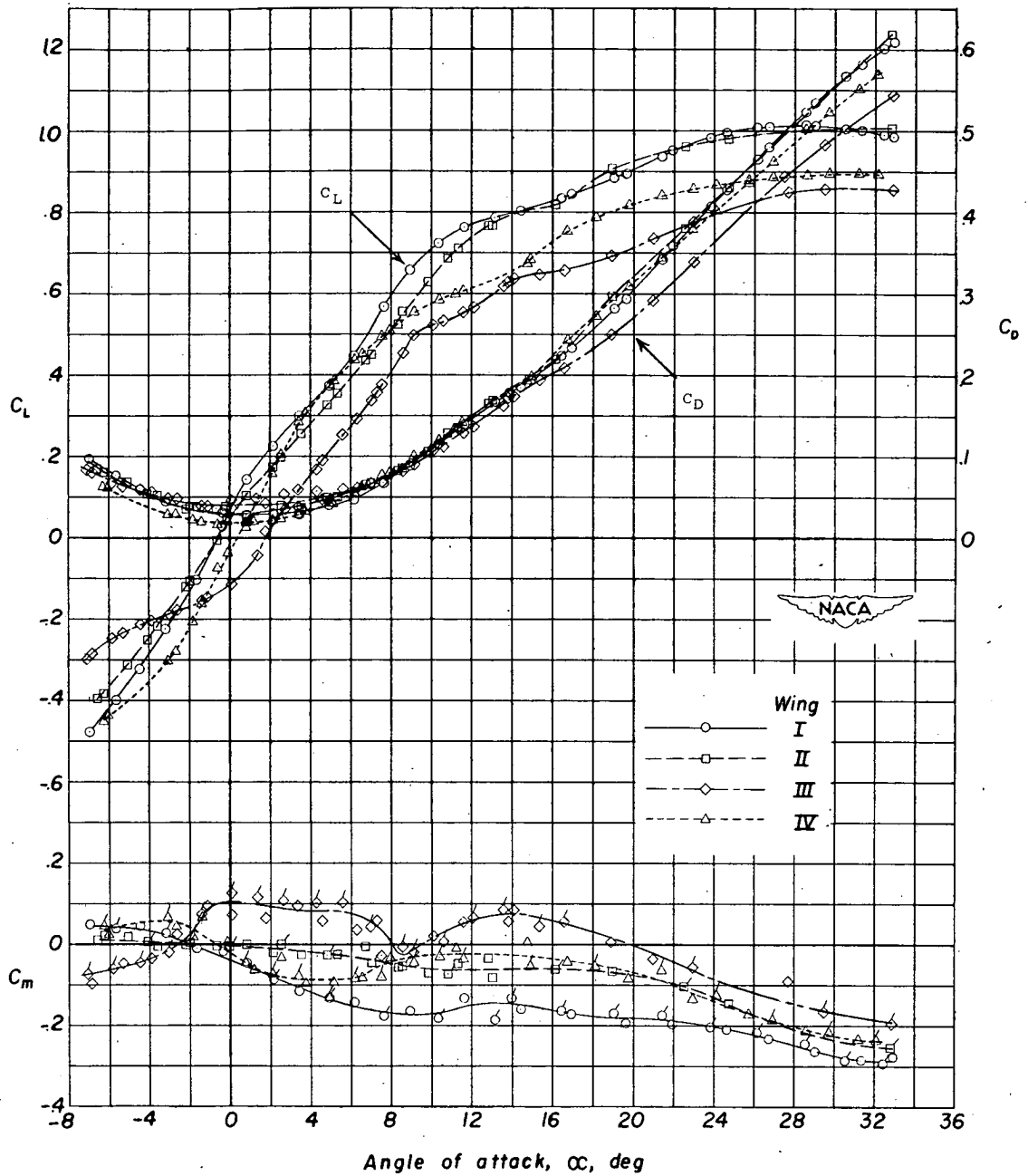
(e) $M = 0.85$.

Figure 11.- Continued.



(f) $M = 0.90$.

Figure 11.- Continued.



(g) $M = 0.95$.

Figure 11.- Continued.

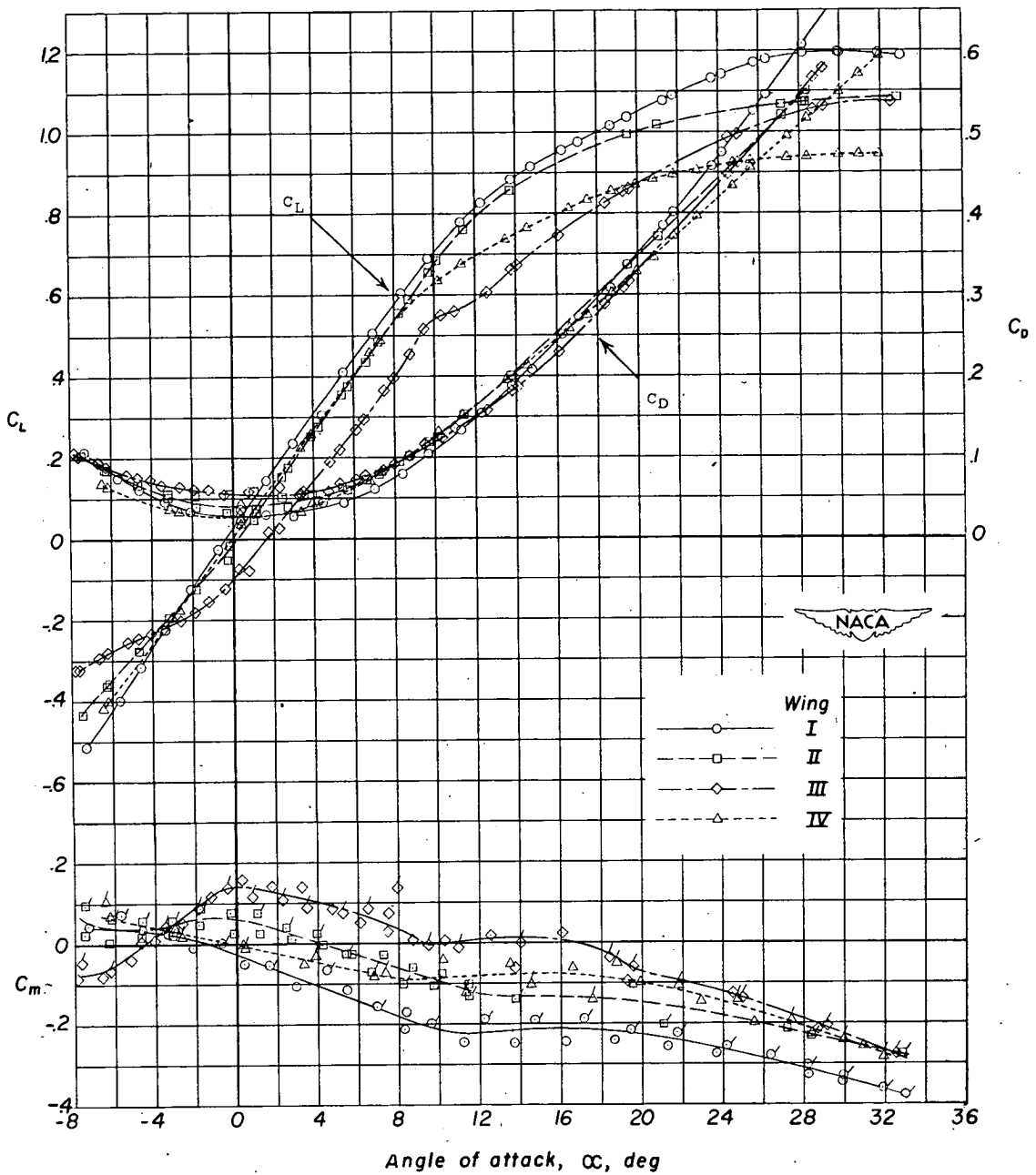
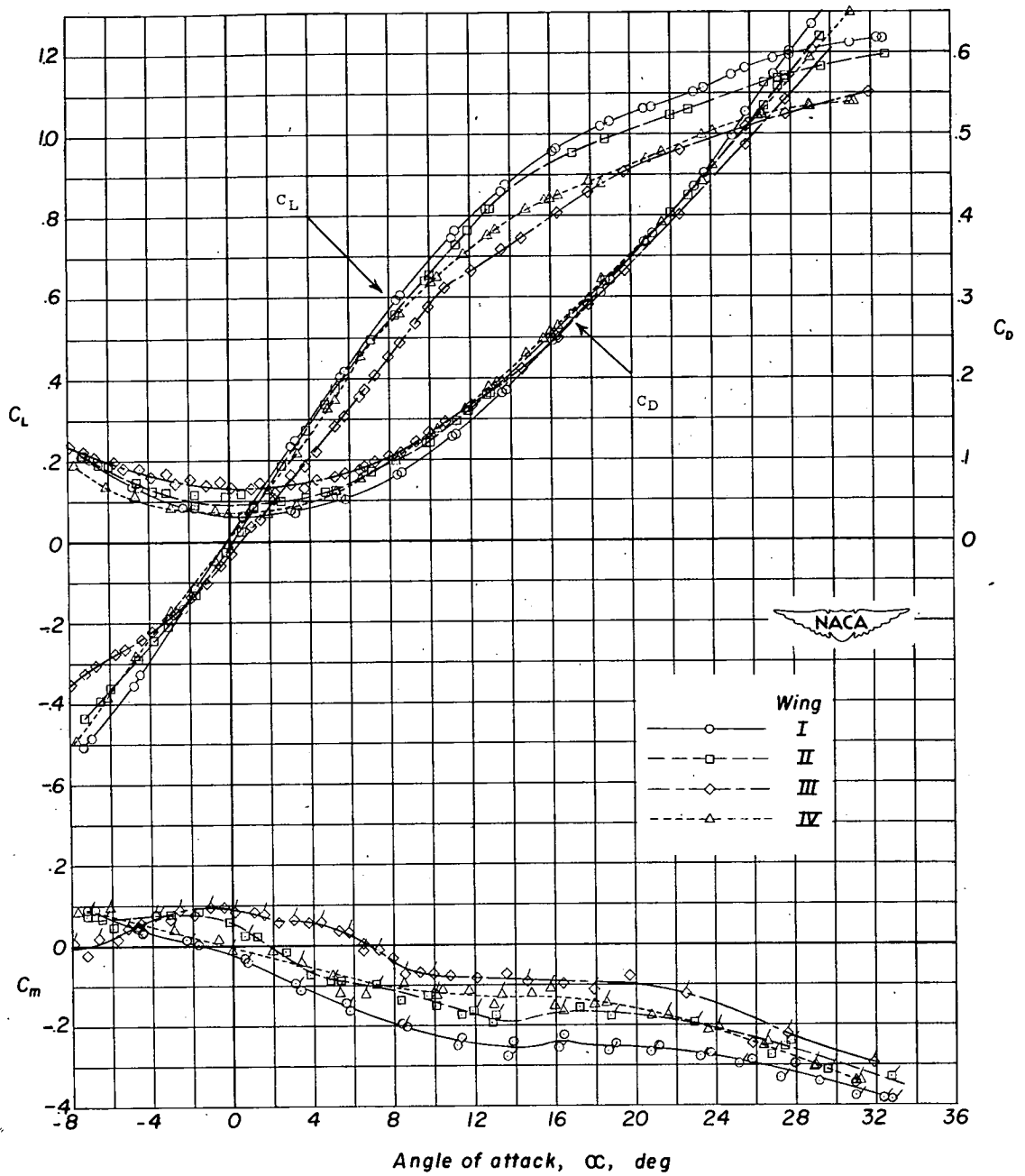
(h) $M = 1.00$.

Figure 11.- Continued.



(i) $M = 1.05$.

Figure 11.- Continued.

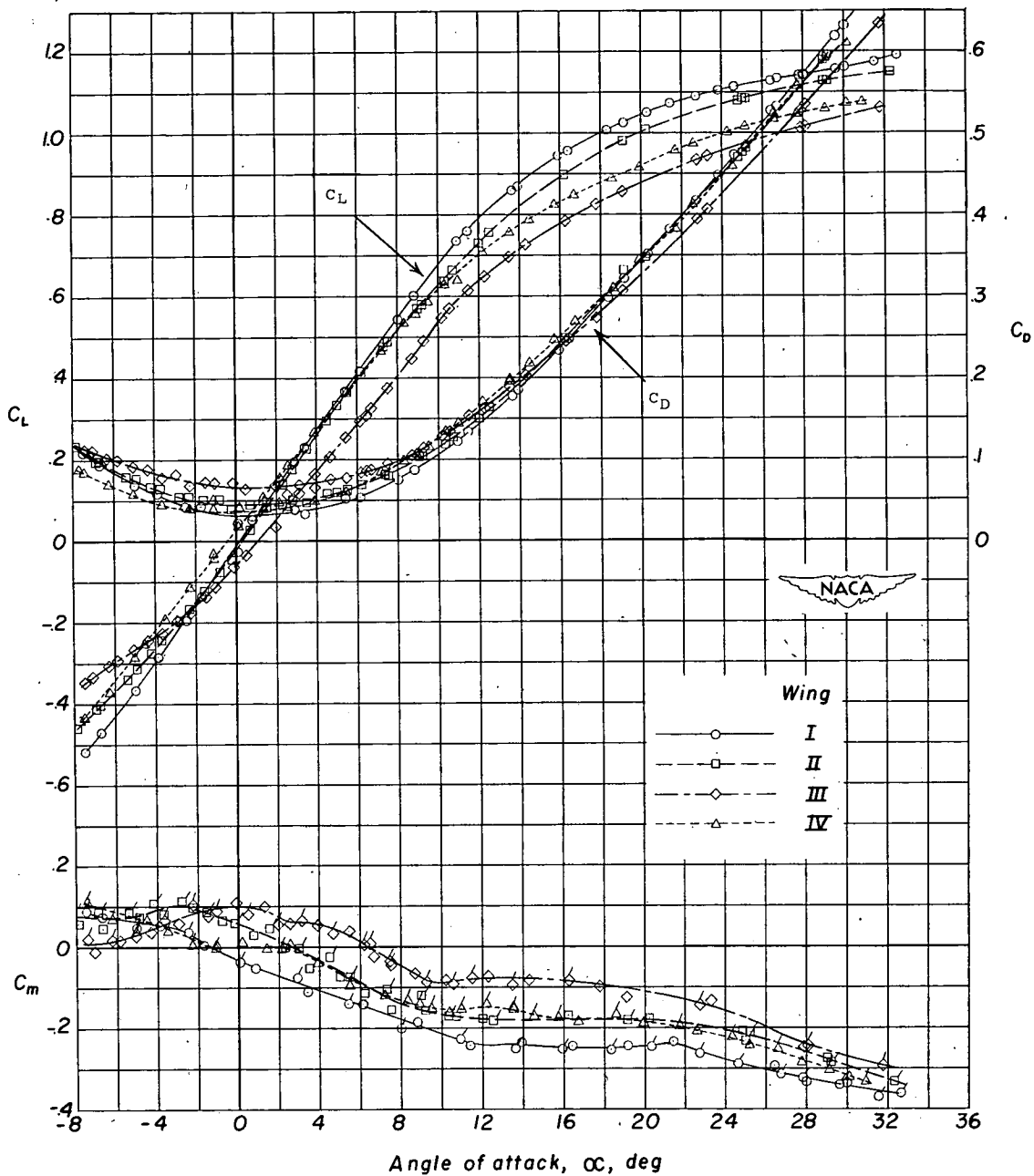
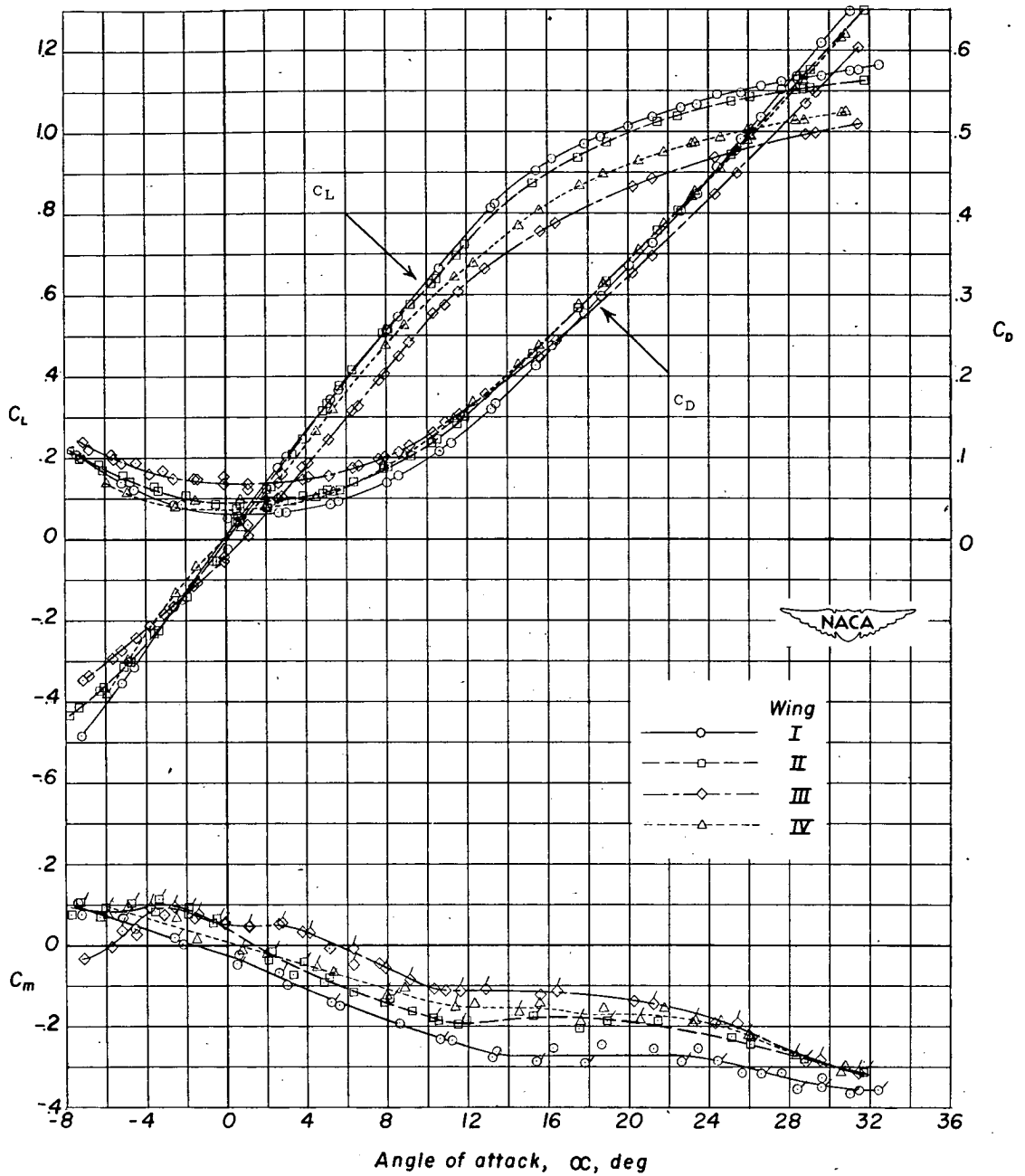
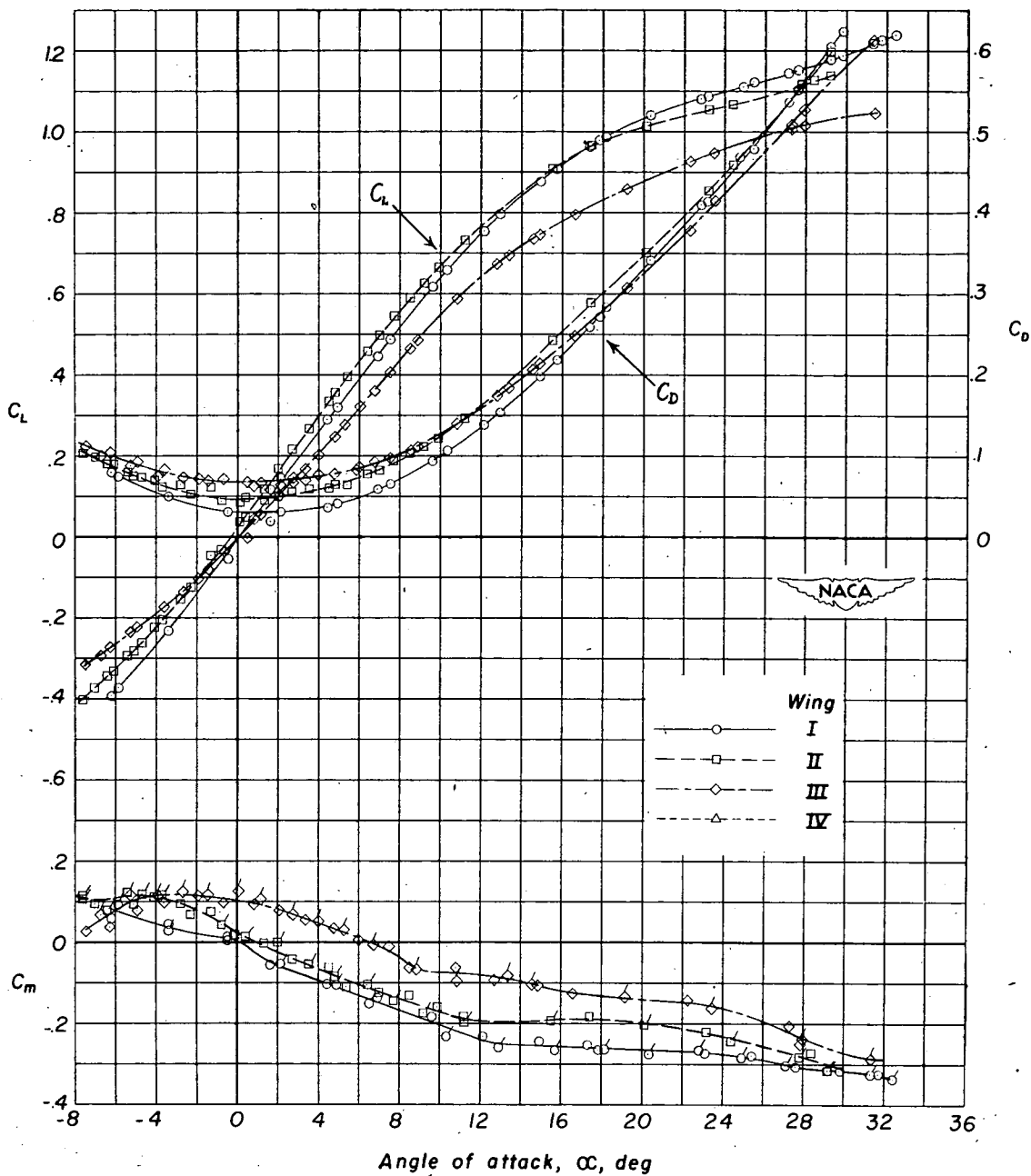
(j) $M = 1.10$.

Figure 11.- Continued.



(k) $M = 1.15$.

Figure 11.- Continued.



(2) $M = 1.175$.

Figure 11.- Concluded.

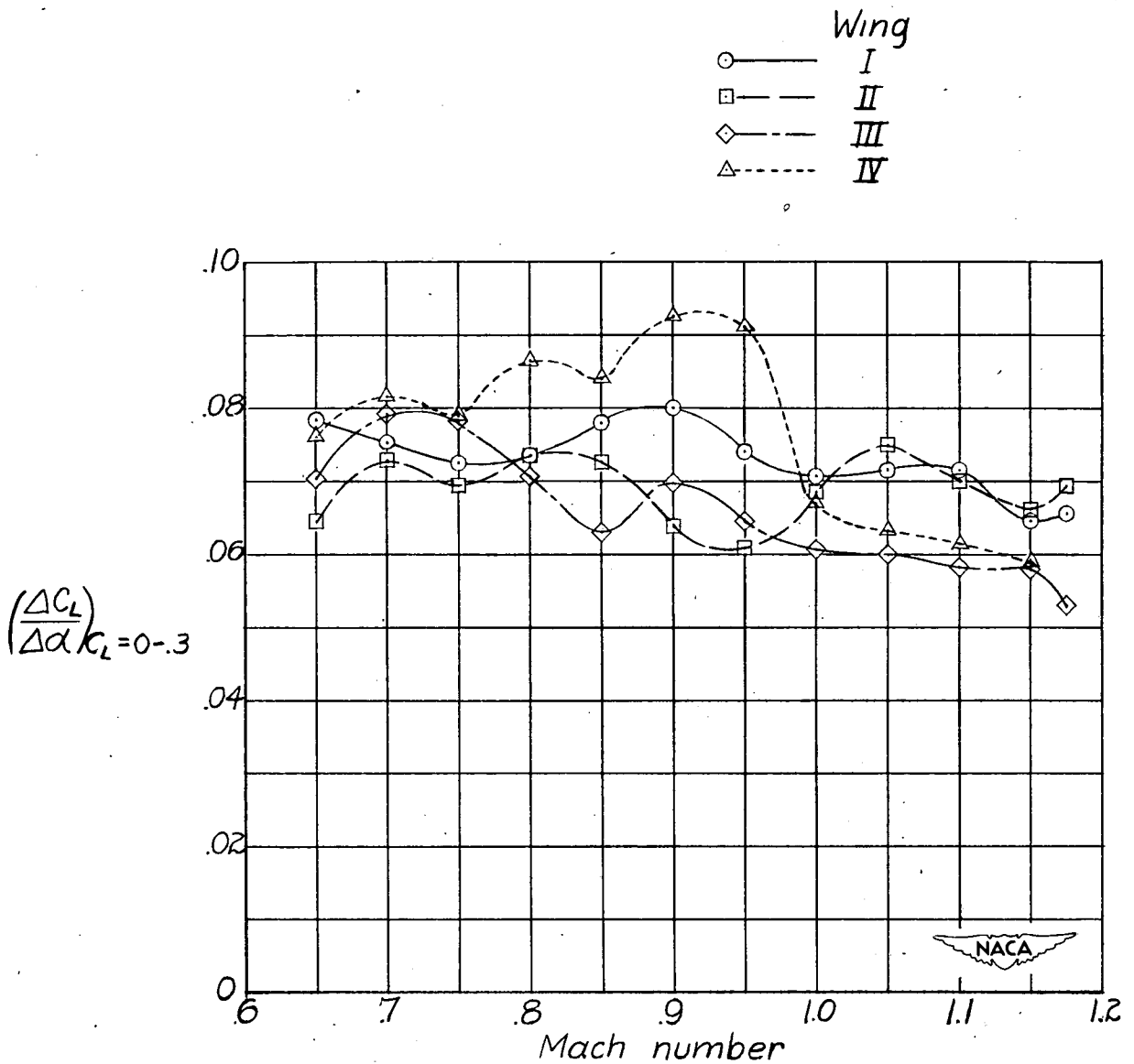


Figure 12.- Variation of lift-curve slope with Mach number for series of wings tested. Lift-curve slopes based on increments of lift and angle of attack between $C_L = 0$ and $C_L = 0.3$. Symbols represent values derived from faired curves of figure 11.

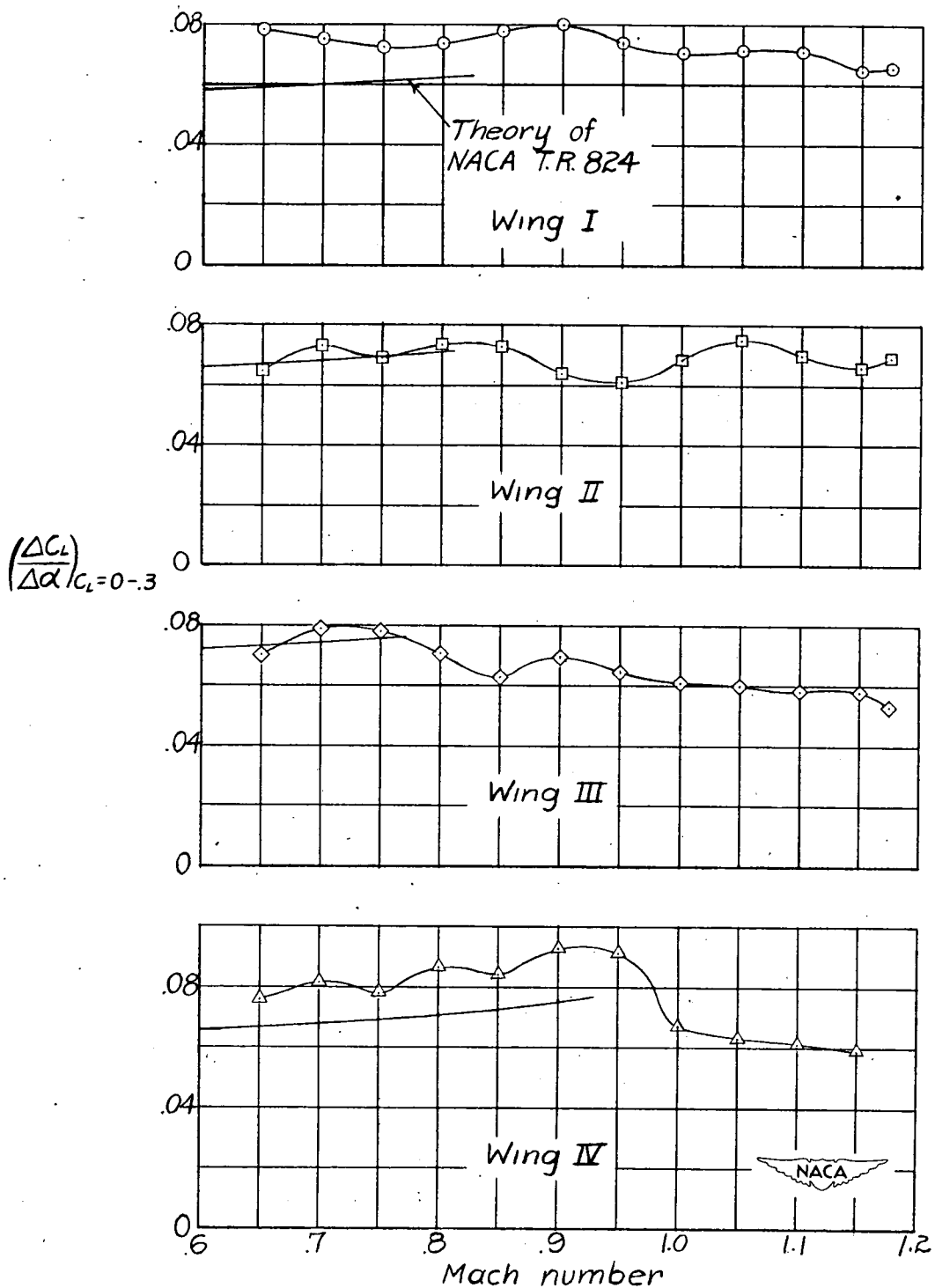


Figure 13.- Comparison between experimental and theoretical lift-curve slopes for wings tested. Experimental lift-curve slopes based on increment of lift between $C_L = 0$ and $C_L = 0.3$. Symbols represent values derived from faired curves of figure 11.

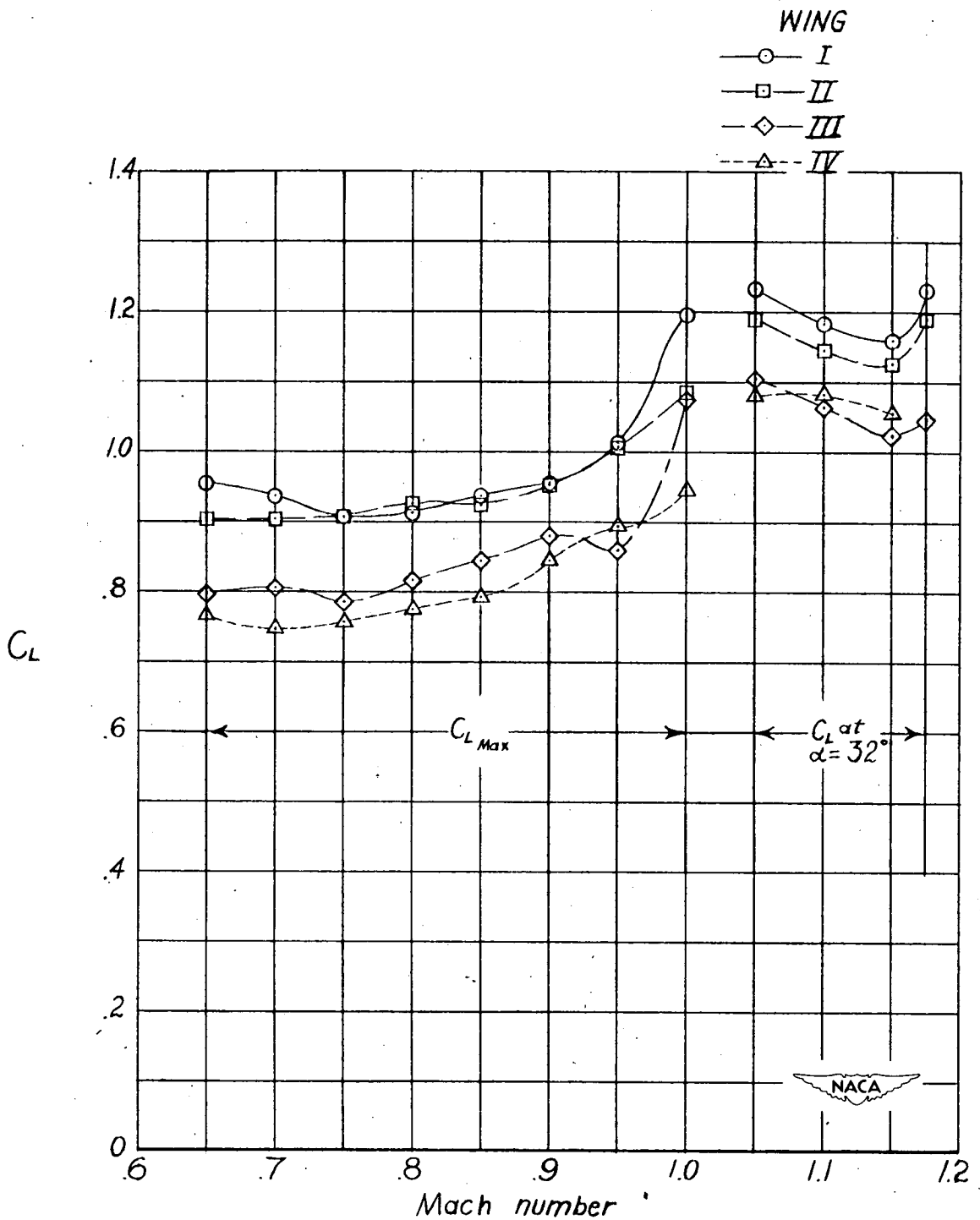


Figure 14.- Variation of maximum measured lift coefficients with Mach number.

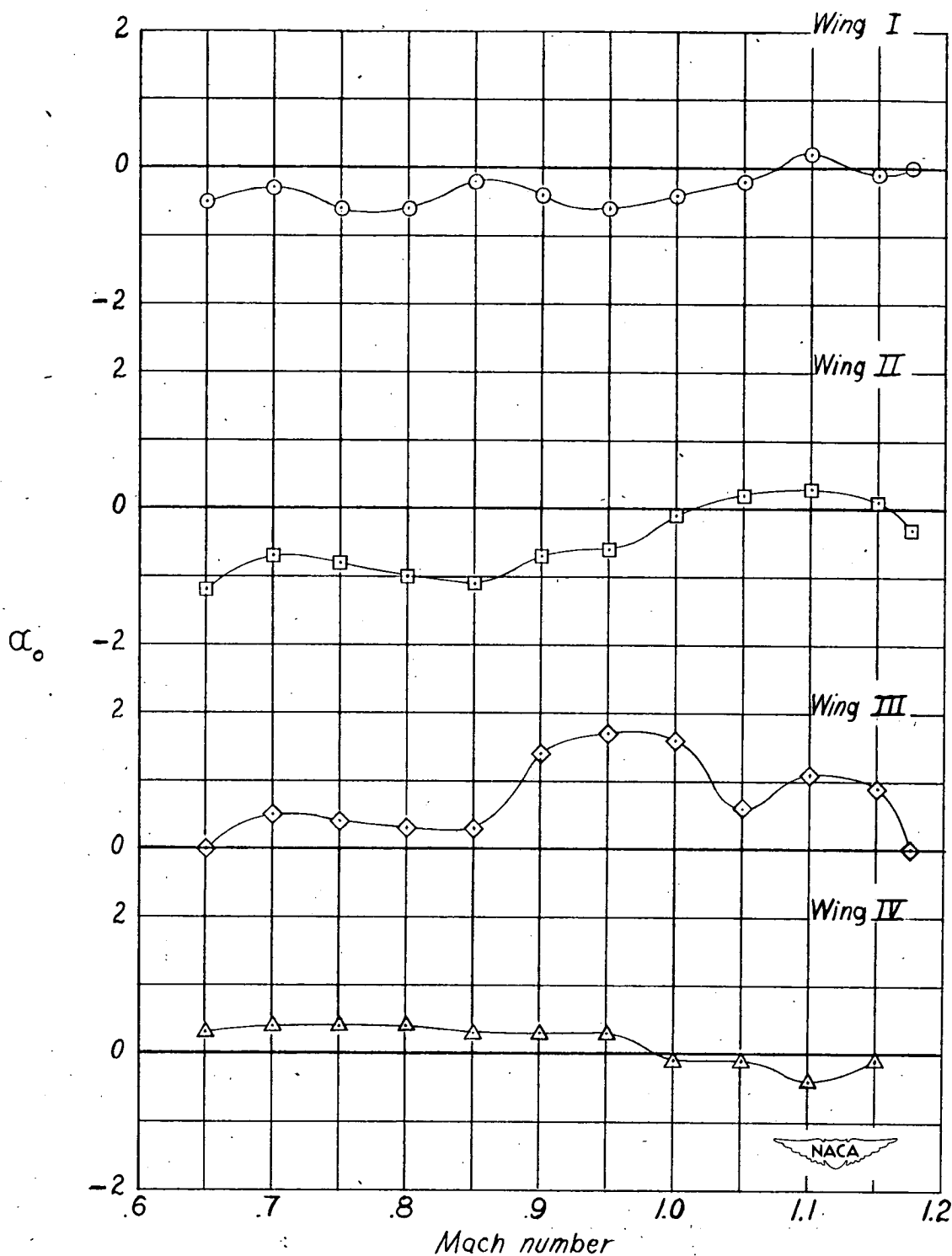


Figure 15.- Angle of attack for zero lift against Mach number. Angles of attack measured with respect to chord lines of root sections of all wings.

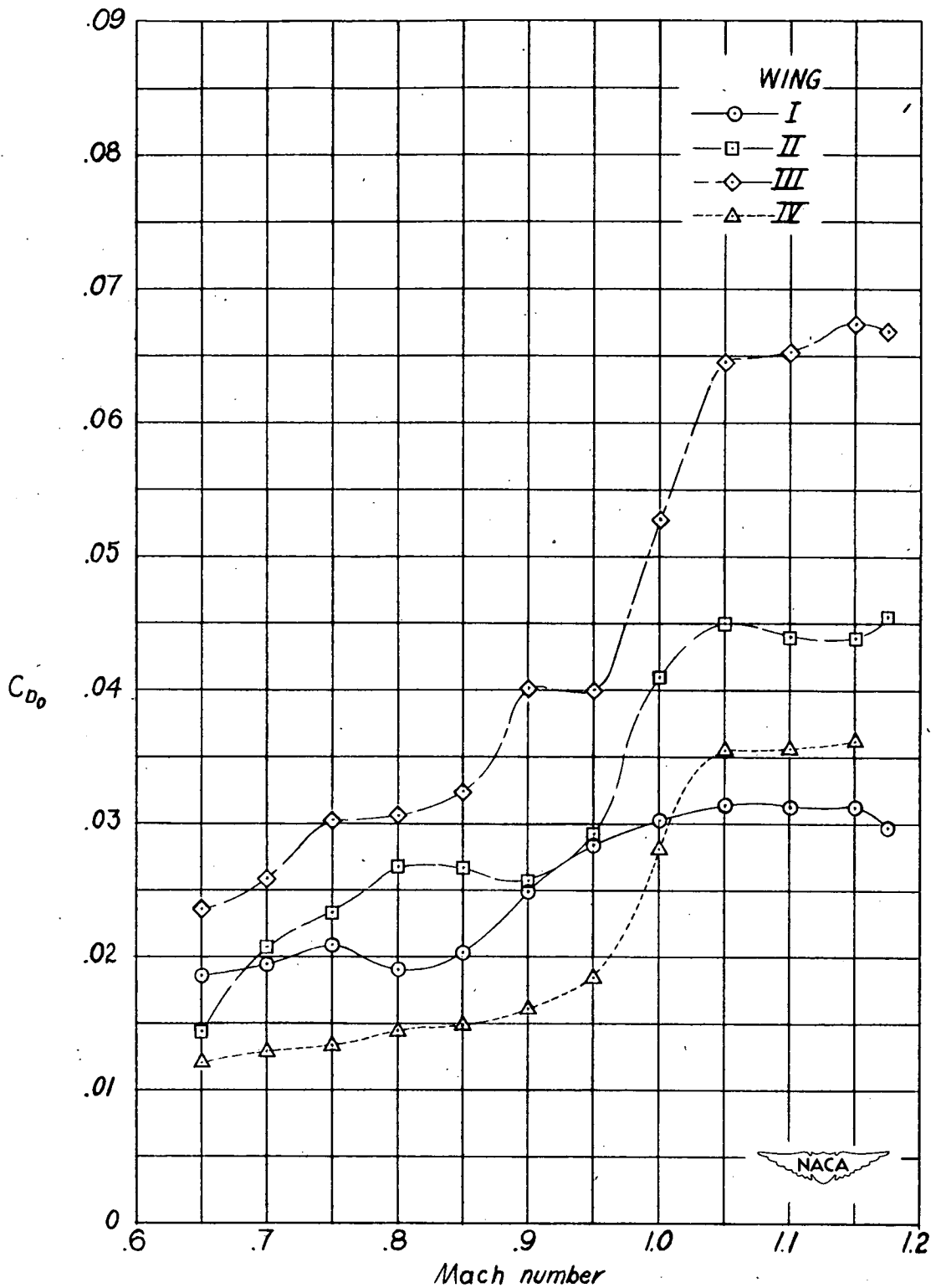


Figure 16.- Variation of drag coefficient at zero lift with Mach number.
Smooth wing condition.

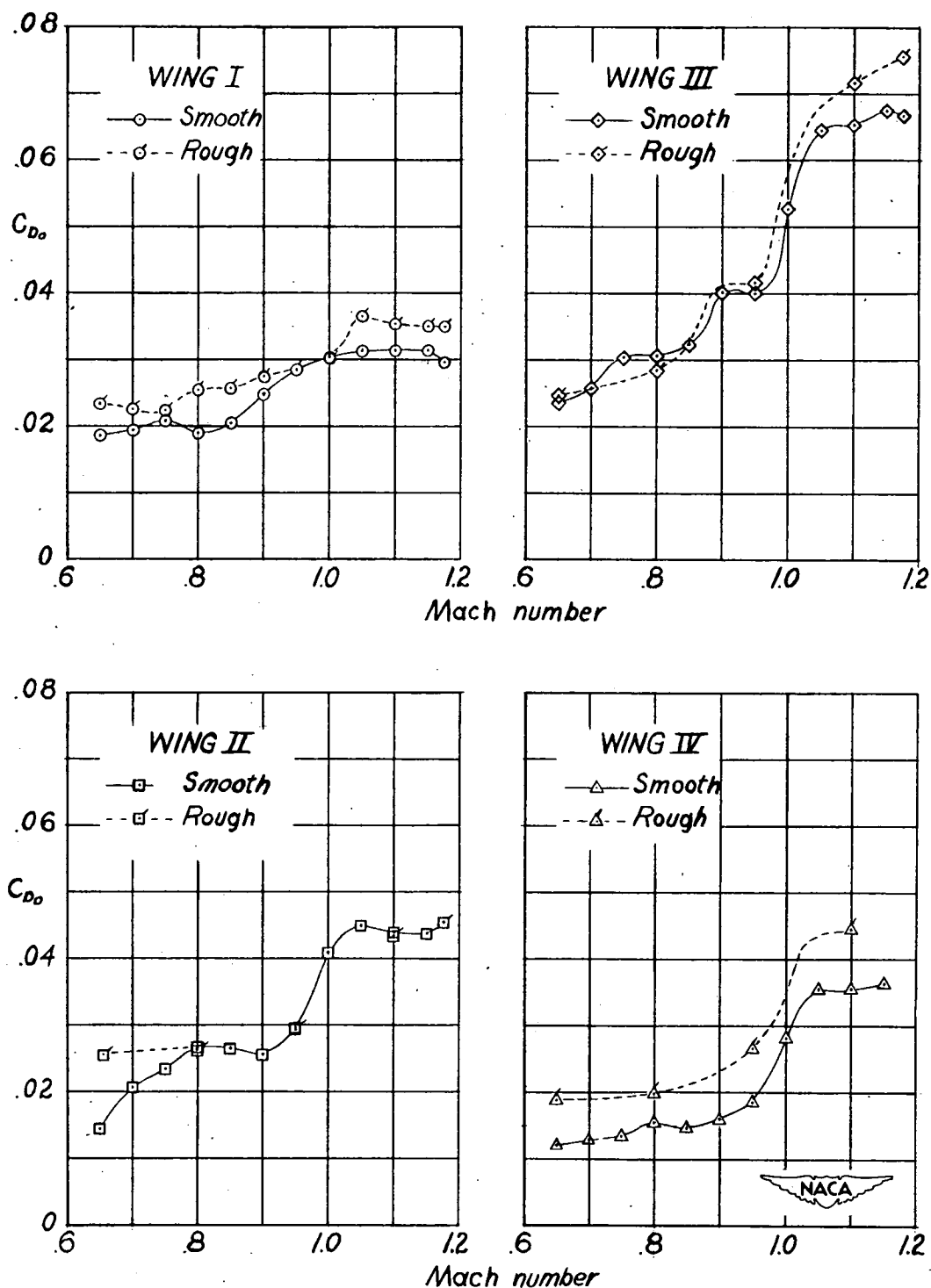


Figure 17.- Effect of surface roughness on zero-lift drag. Roughness consisted of 0.003- to 0.005-inch particles on both surfaces from 0 to 5 percent wing chord.

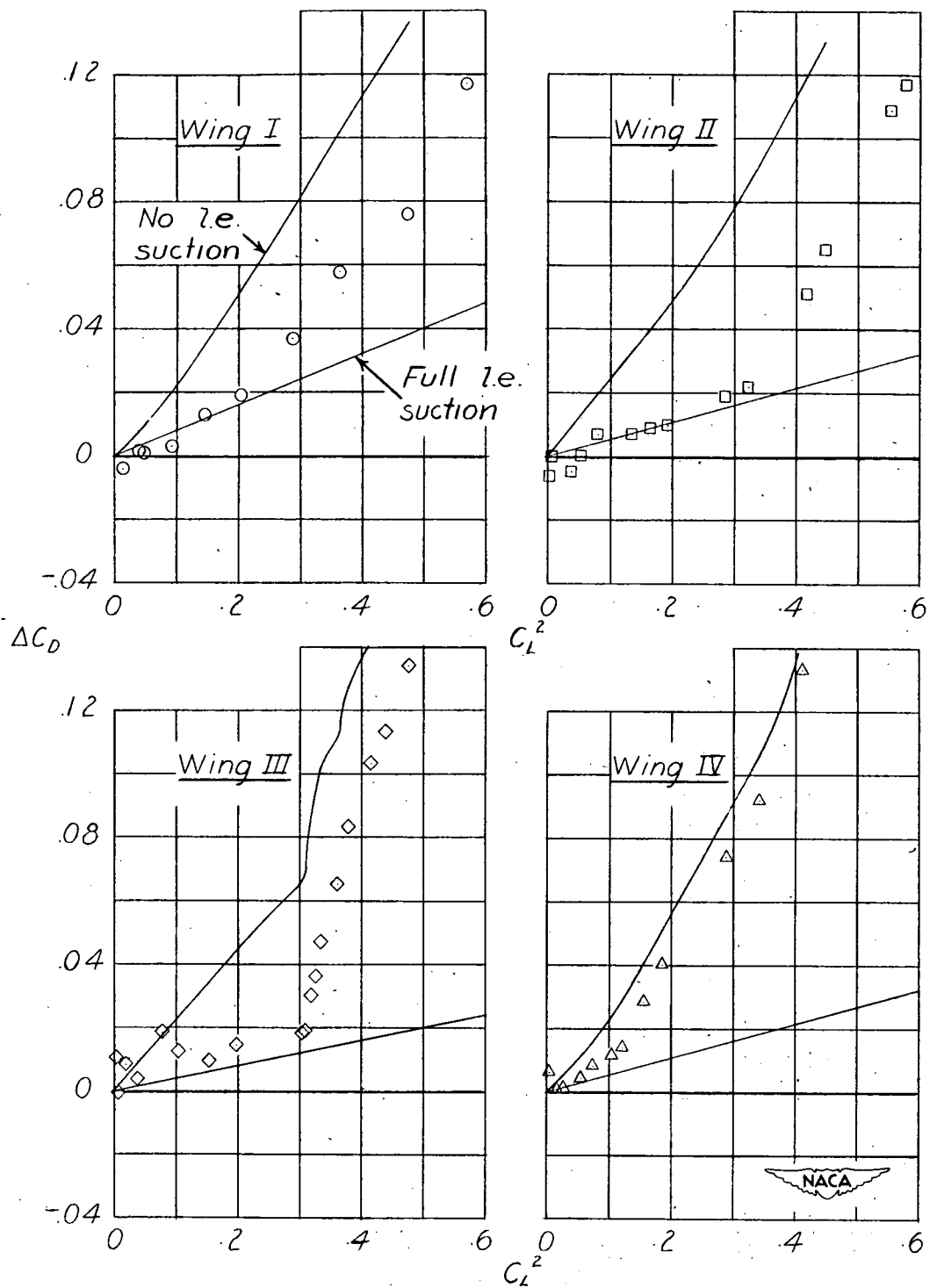
(a) $M = 0.70$.

Figure 18.- Increment of drag coefficient due to lift against square of lift coefficient at three representative Mach numbers.

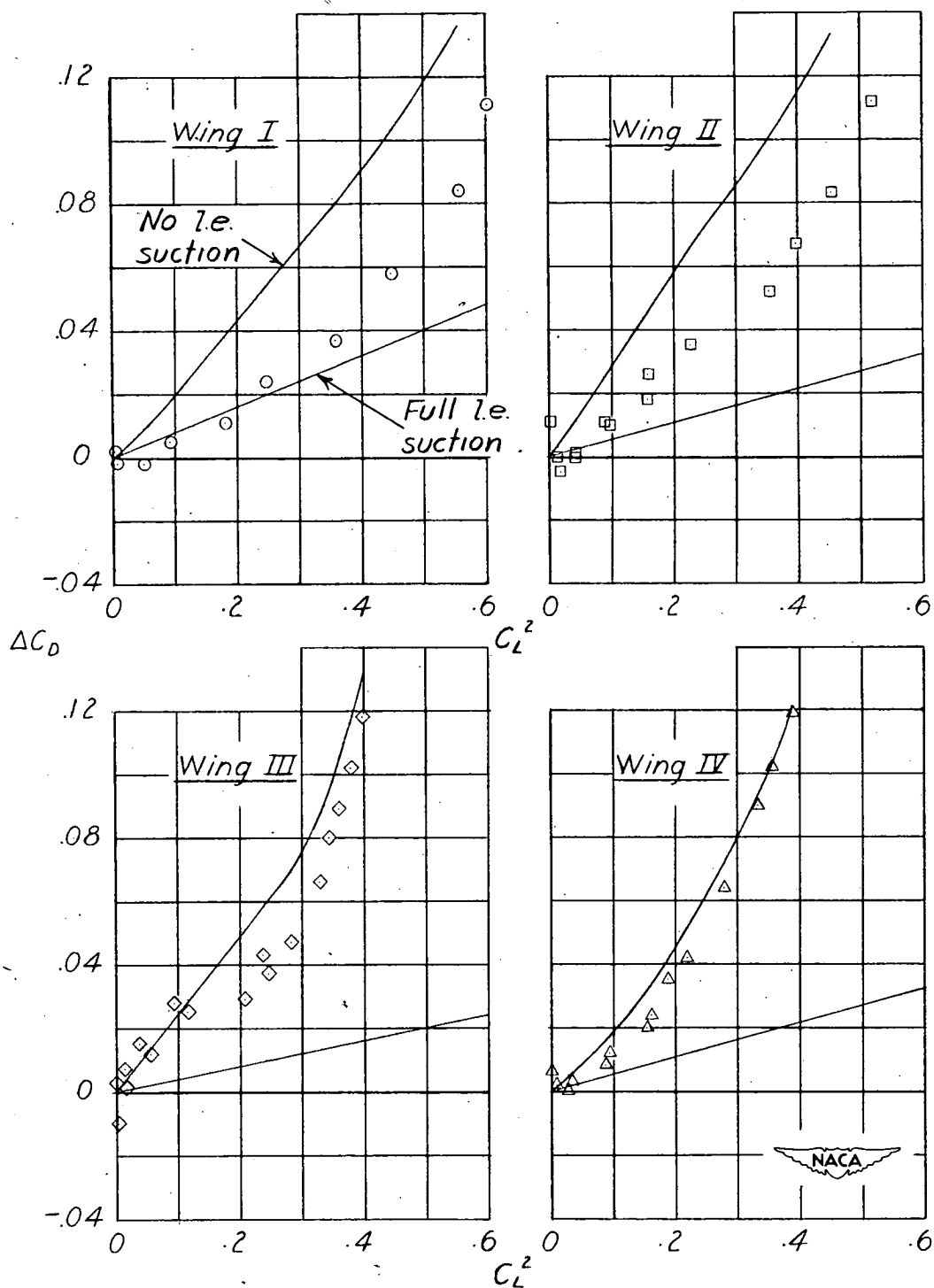
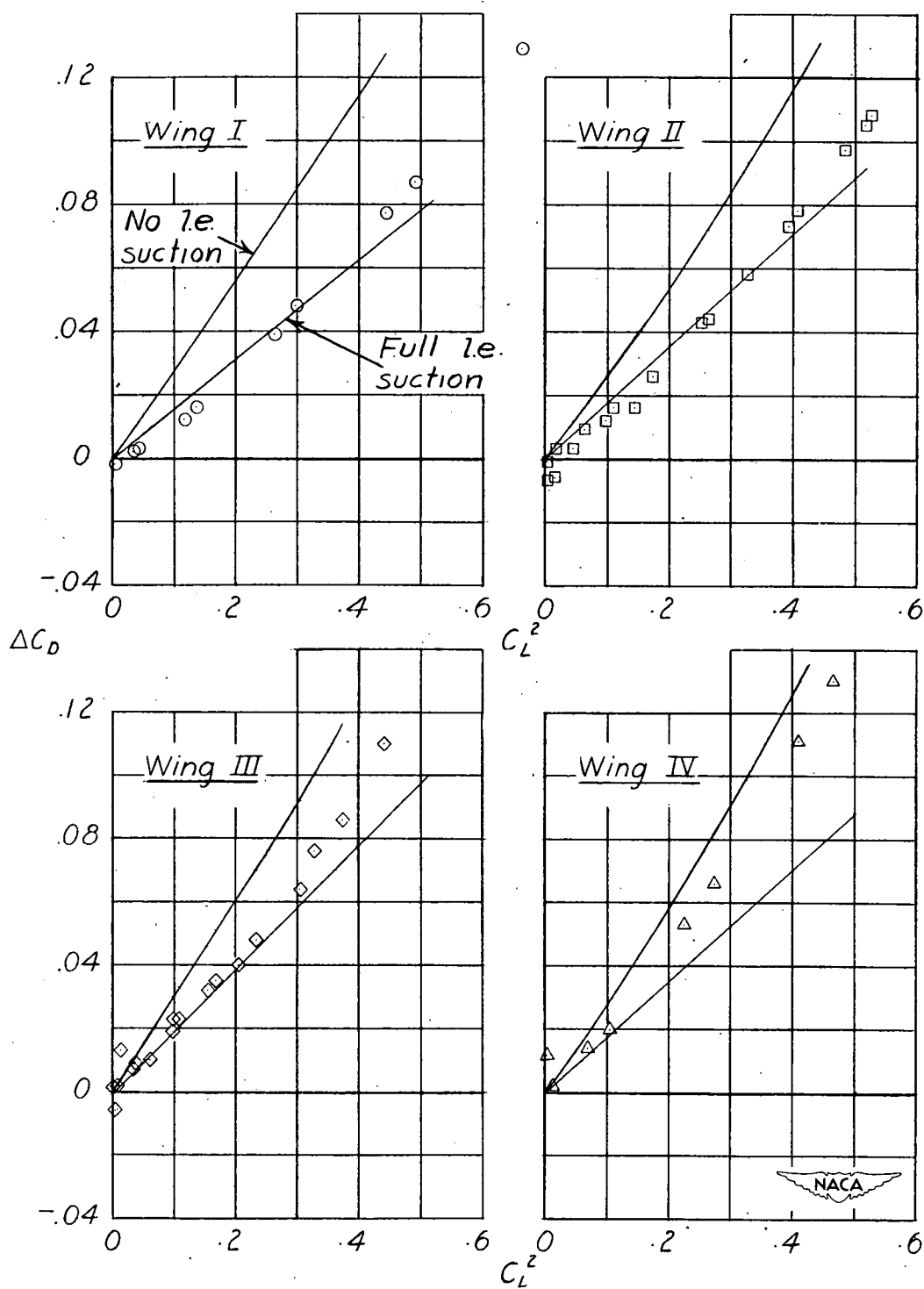
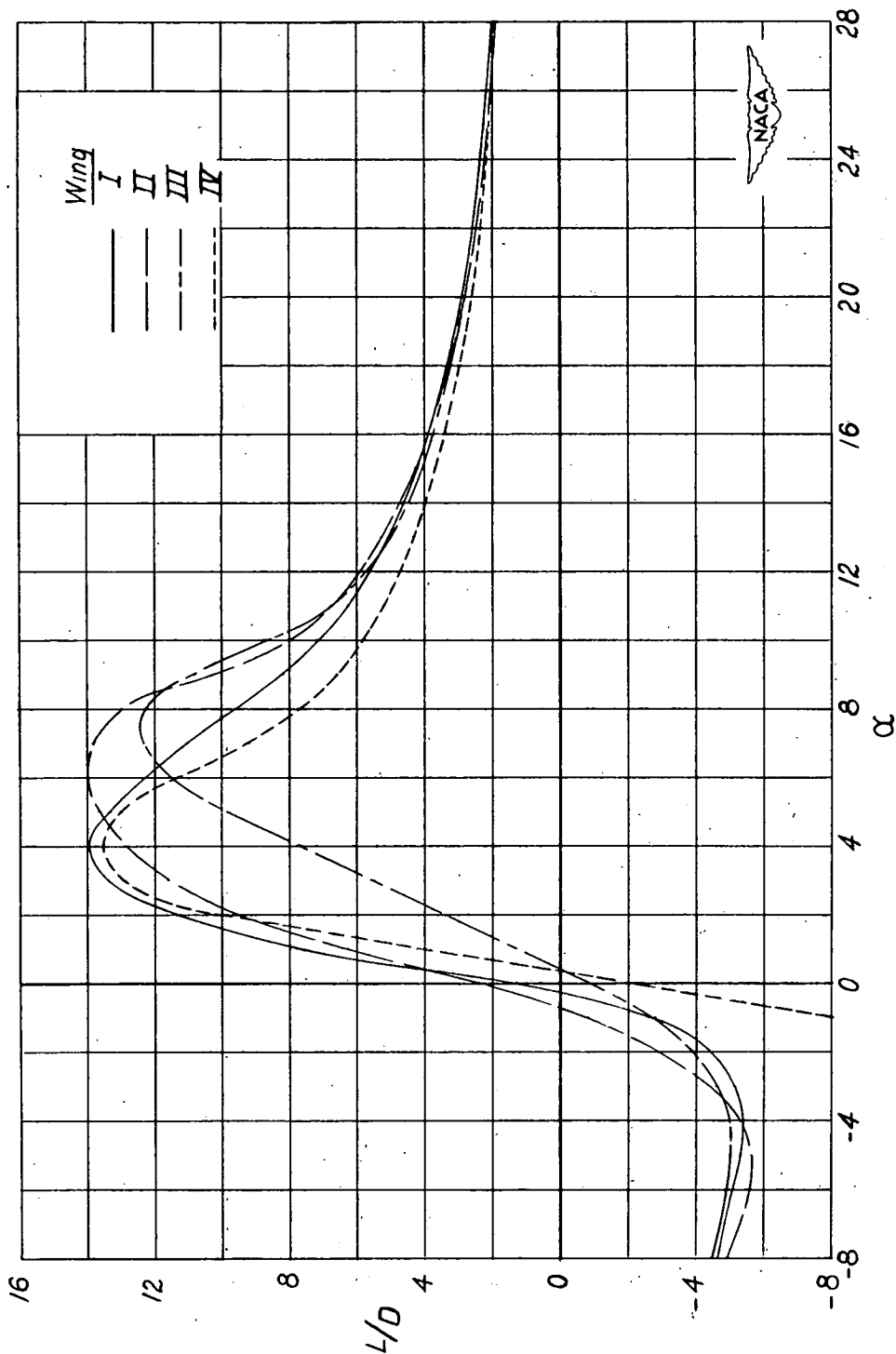
(b) $M = 0.90$.

Figure 18.- Continued.



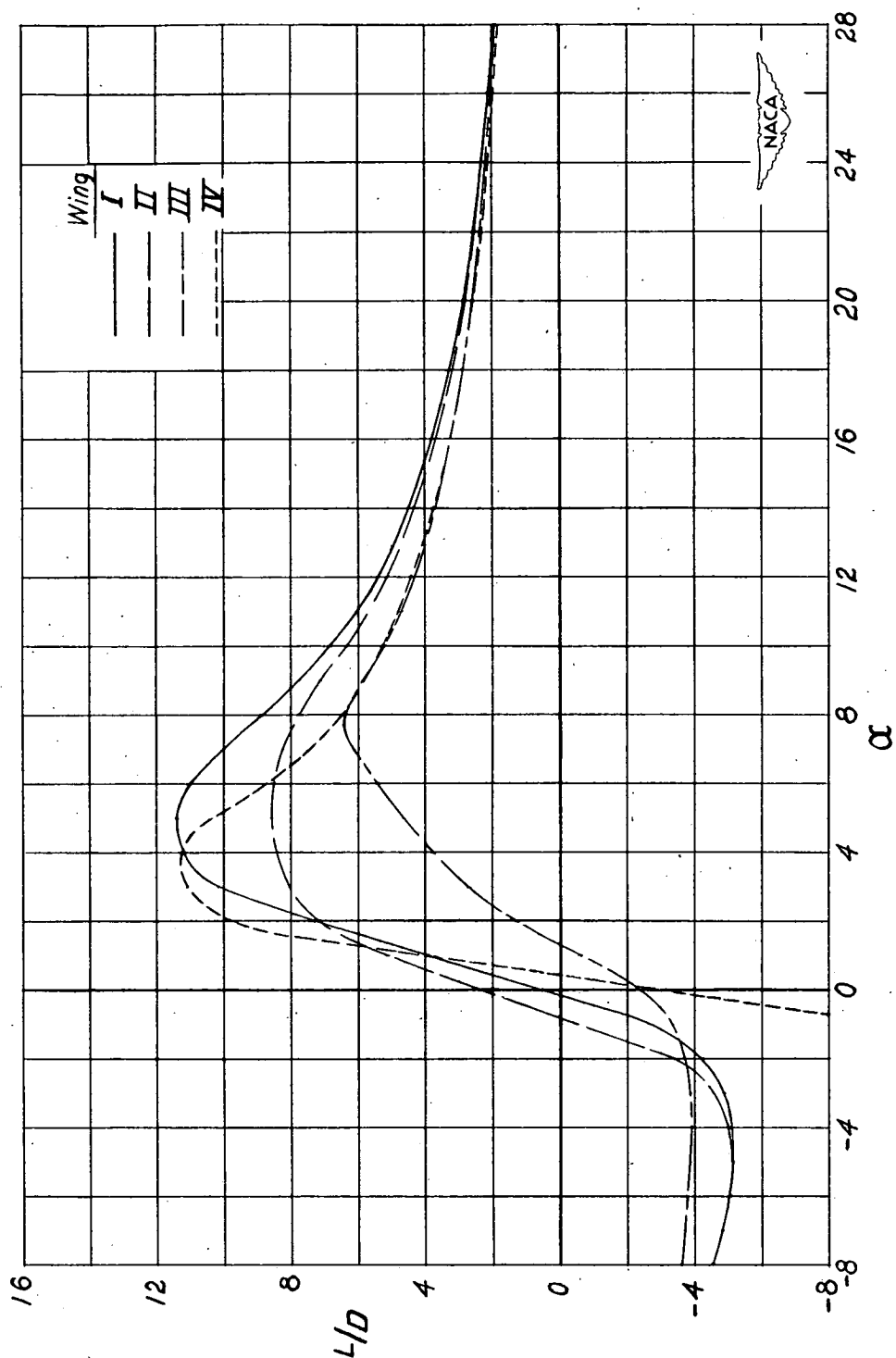
(c) $M = 1.15$.

Figure 18.- Concluded.



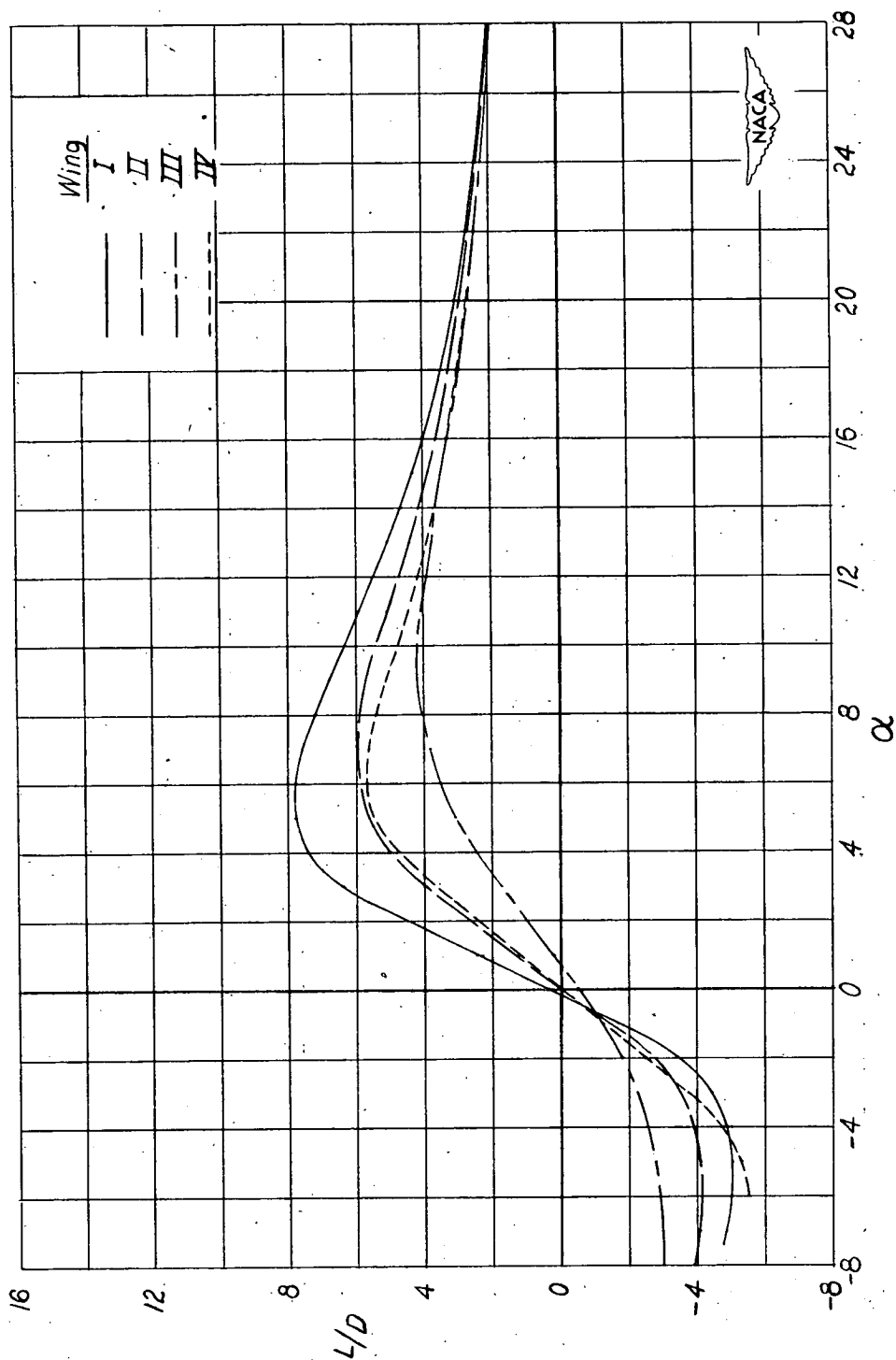
(a) $M = 0.70$.

Figure 19.- Variation of lift-drag ratio with angle of attack at three representative Mach numbers.



(b) $M = 0.90$.

Figure 19.- Continued.



(c) $M = 1.15$.

Figure 19.- Concluded.

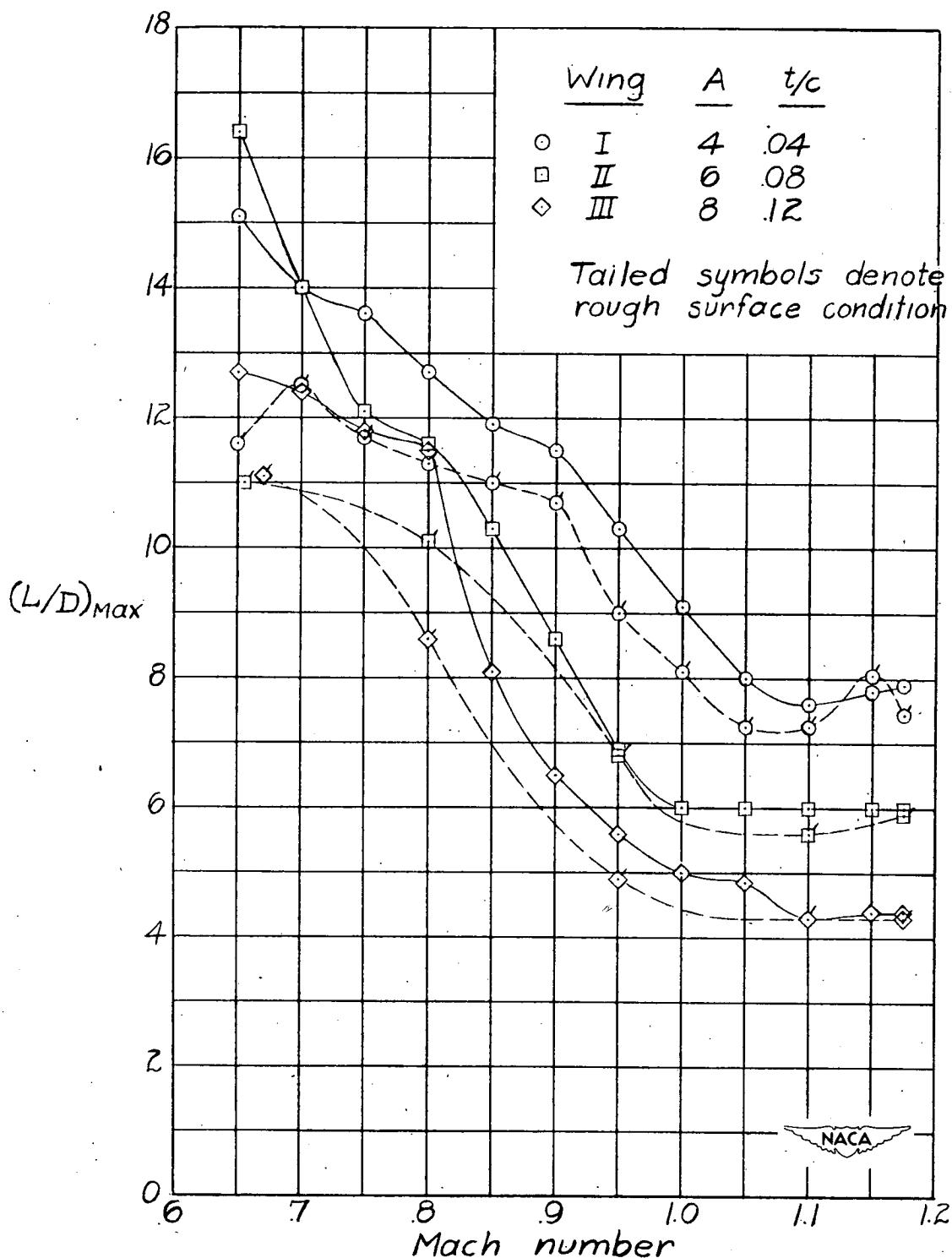


Figure 20.- Variation of maximum lift-drag ratio with Mach number in both smooth and rough surface conditions for the three cambered and twisted wings tested.

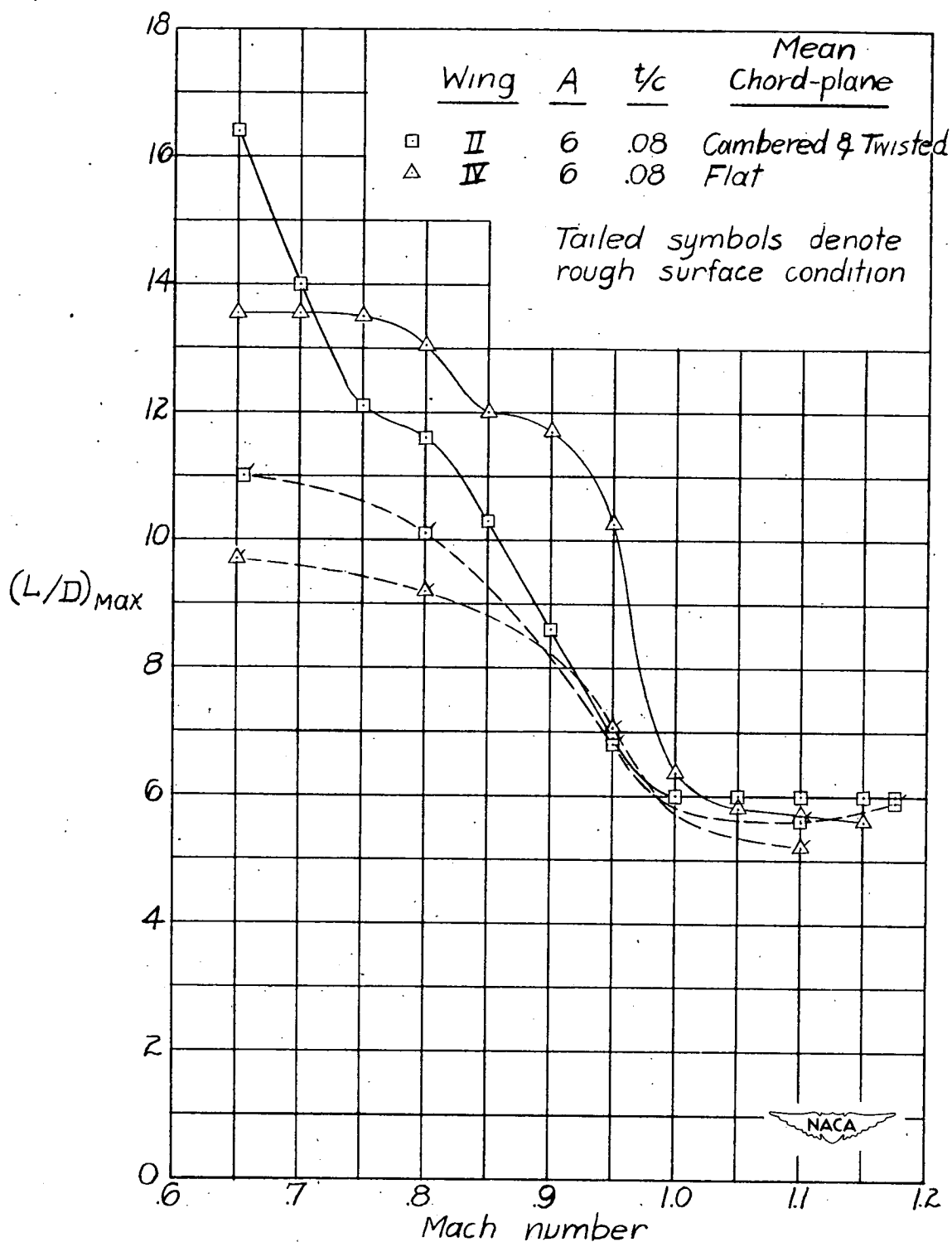


Figure 21.- Effect of camber and twist on the maximum lift-drag ratio of a 45° sweptback wing of taper ratio 0.25. Cambered and twisted wing design lift coefficient = 0.3 at $M = 0.90$.

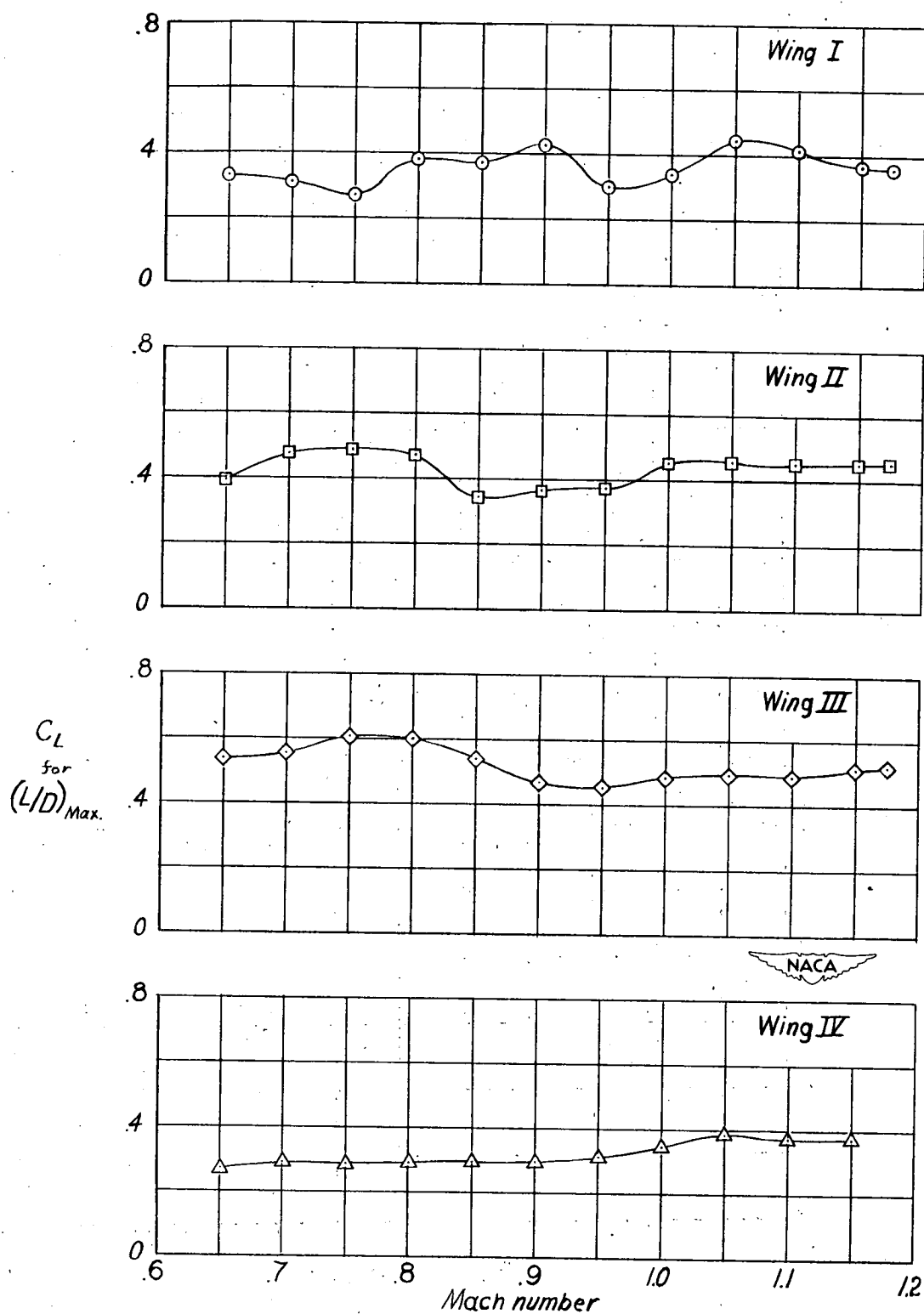


Figure 22.- Variation of lift coefficient for maximum L/D with Mach number.

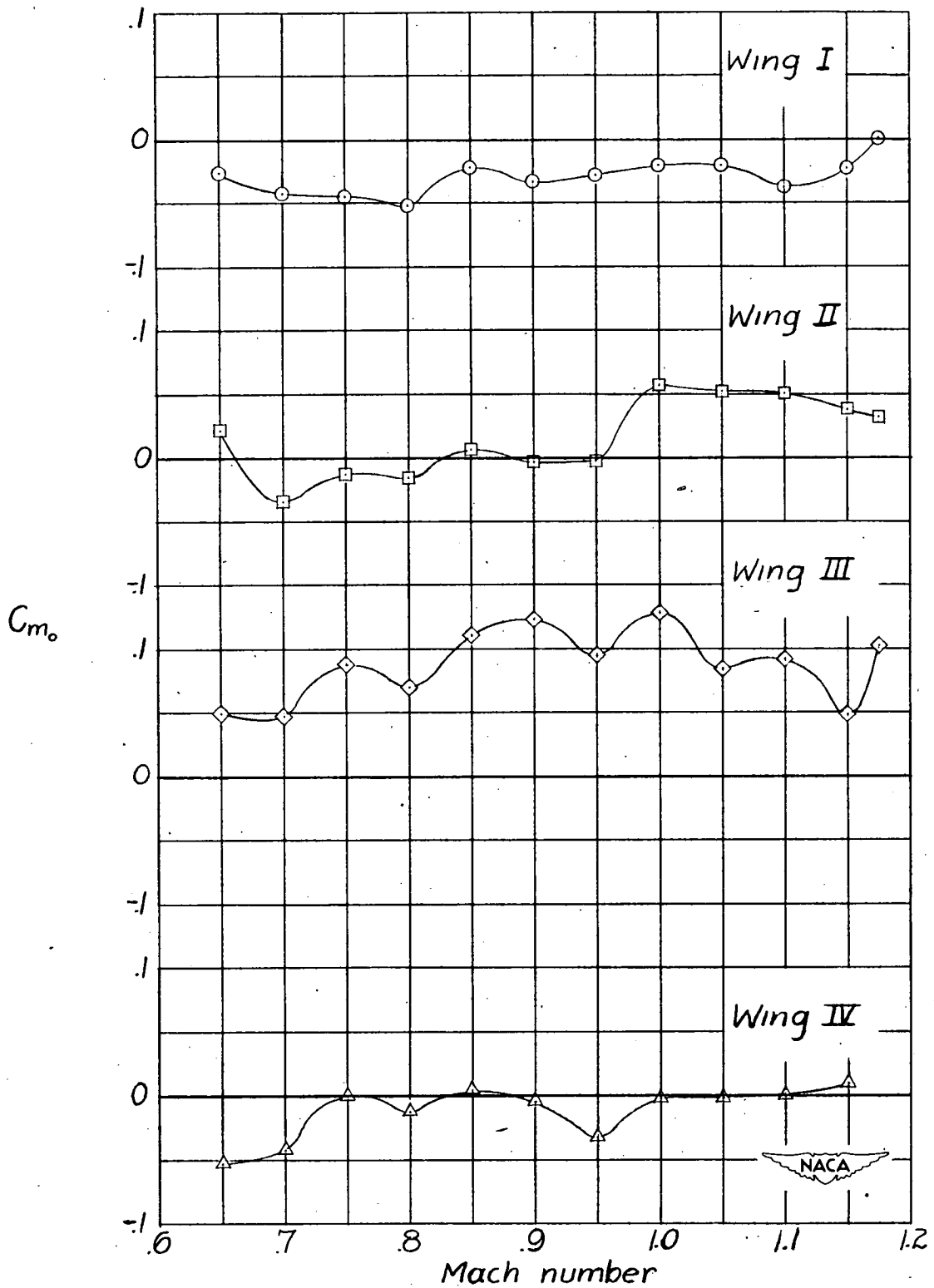


Figure 23.- Zero-lift pitching-moment coefficient against Mach number.

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